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GAS TURBINES AND JET PROPULSION

Gas Turbines and Jet Propulsion

BY G. GEOFFREY SMITH, M.B.E.

REVISED AND ENLARGED BY
F. C. SHEFFIELD

With 342 photographs and drawings in
the text and 4 illustrations inset

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Dedicated to the Royal Aeronautical Society as a small tribute to their constant endeavour to advance aeronautical science and engineering from 1866 *to the present day*

Foreword

BY SIR HAROLD ROXBEE COX, PH.D., D.I.C., D.SC.,
M.I.MECH.E., F.R.AE.S., F.INST.F., F.I.AE.S.

President, Royal Aeronautical Society, 1947–1949
Director, National Gas Turbine Establishment, 1946–1948

THE most appropriate monument to a man's memory is often one he has created himself, one that enshrines his outlook and his energies and his enthusiasms, one like this book, which bids fair to stand the test of time as effectively as a monument of stone. I feel, in writing this foreword to the sixth edition of his book, honoured to have, in this modest way, an opportunity to pay a humble tribute to the contribution which Geoffrey Smith made in the last decade of his life to the progress of the gas turbine by preparing men's minds to receive a revolutionary change in methods of propulsion.

In writing this foreword I want to repeat myself. I want to write again some of what I wrote for the fifth edition, and particularly to record my appreciation of the compliment which Geoffrey Smith paid me in asking me thus to introduce his work.

During the war I was engaged with the teams concerned with the intensive research and development of the gas turbine for aircraft. Our work was necessarily secret, and it was not possible to discuss with many engineers interested in the gas turbine the nature of our results nor the implications of our work. Fortunately, Geoffrey Smith was one of the relatively few independent engineers who had examined the possibilities of the gas turbine for aircraft on the basis of the published data, and he had developed a belief in its future at a time when most of those around him were extremely sceptical about its possibilities. He felt very strongly that the minds of technical men should be prepared for acceptance of the gas turbine for aircraft, and, consequently, he began to write articles explaining the principle of the gas turbine and what it could do for the aeroplane.

At an early stage he realized that the advantages of the gas turbine were of two kinds—those inherent in itself, and those which it conferred upon the aeroplane. In other words, he tried to make clear from the outset that airframe design itself would benefit from its integration with the new form of power plant.

Naturally, because of his work for the Government during the war, he knew that many of his friends, including myself, were working in the gas turbine field, and he outlined his views to us. We should very much have liked to outline our views to him! The need for strict secrecy meant, however, that we were unable to give this valiant exponent of our cause any of the very convincing ammunition in our possession, which would have helped him in his task of preparing the way for general acceptance of this new type of power plant. However, he managed very well without our help.

Technical change is apt to take place in three stages—the development and trial of an idea, a revolution, a consolidation. When writing four years ago, I said that the second stage, in the case of the aircraft gas turbine, was well advanced. Now I think I would say we are almost at the beginning of the third. It is amazing that since 1941, when the idea had been translated into hardware and first demonstrated here in flight, so remarkable a change should have taken place in aircraft propulsion. The Air Forces of the world have accepted the gas turbine, and while they still have numbers of piston-engined aircraft, they are planning all the time to exchange as many as possible for turbine-engined aircraft. The revolution must develop more slowly in the civil aircraft world because civil aircraft cannot economically be replaced immediately a more attractive technical proposition appears. It is certain, however, that the fastest of them will be jet-propelled and those of more modest performance driven by propeller turbines, while only light aircraft and freighter types can look forward for some years to employing what many still refer to as the conventional means of propulsion.

In reliability the gas turbine is forging steadily ahead. There is no reason to believe that its maintenance costs will be higher than those of the piston engine. Its fuel consumption is being steadily improved. Soon *it* will be regarded by everyone as the conventional means of propulsion and the revolution will be complete.

The rate of increase of technical knowledge is something to be a little frightened of. Man's knowledge has outrun his wisdom. The stimulus which drives us to greater and greater speeds—and that is what the gas turbine is giving us in the air—is not necessarily ' a good thing,' but if there is to be an ever-accelerating race against time, then it is well to be out in front. There is no doubt that the work of British gas turbine engineers has given us a lead in this race. To have been associated with the great work which has led to our being in this position is an experience I would not have missed for worlds. And that the story of this work, which has not yet ended, should have found so gifted a chronicler is something for which we should all be grateful.

Now he is gone. His natural successor is an old associate, Mr. Sheffield, whom I often met in his company. Mr. Sheffield is an able technologist and a clear expositor, and it is most fortunate that he is available to carry this work to its sixth edition. May many more be required of him.

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Abbreviations

abs	Absolute	I.C.	Internal combustion
A.C.	Alternating current	I.L.S.	Instrument landing system
A.C.U.	Altitude control unit	in	Inch
a.m.p.g.	Air miles per gallon	i-p	Intermediate-pressure
atm	Atmosphere	°K	Degree Kelvin
A.T.O.	Assisted take-off	kc/s	Kilocycles per second
a.u.w.	All-up weight	kg	Kilogram
b.h.p.	Brake horse-power	kt	Knot
B.Th.U.	British Thermal Unit	kW	Kilowatt
°C	Degree Centigrade	kWh	Kilowatt-hour
cal	Small calorie	l	Litre
c.g.	Centre of gravity	lb	Pound
cm	Centimetre	lb/hr/lb	Lb fuel per hour per lb thrust
c/s	Cycles per second	L/D	Lift:drag ratio
cs	Centistokes	l-p	Low-pressure
dB	Decibel	m	Metre
D.C.	Direct current	min	Minute
D.I.	Daily inspection	mm	Millimetre
e.h.p.	Equivalent horse-power	m-p	Medium-pressure
°F	Degree Fahrenheit	m.p.h.	Miles per hour
ft	Foot	P.B.O.	Pool burning oil
ft-lb	Foot-pound	r.p.m.	Revolutions per minute
g	Gravity: 32·18 ft/sec^2	sec	Second
gal	Imperial gallon	s.h.p.	Shaft horse-power
G.C.A.	Ground-controlled approach	s.l.s.	Sea level static
g.cal	Gram-calorie	S.T.	Static thrust
g.h.p.	Gas horse-power	T.A.S.	True air speed
Hg	Mercury	t.h.p.	Thrust horse-power
h-p	High-pressure	t.o.h.p.	Take-off horse-power
h.p.	Horse-power	V	Velocity
hr	Hour	yd	Yard

Equivalents

1 in	2·54 cm
1 ft	0·3048 m
1 yd	0·9144 m
1 mile	1·6093 km
1 in²	6·4516 cm²
1 ft²	0·0929 m²
1 yd²	0·8361 m²
1 mile²	2·5900 km²
1 in³	16·387 cm³
1 ft³	0·0283 m³
1 yd³	0·7645 m³
1 pt	0·568 l
1 qt	1·136 l
1 gal	4·546
1 U.S. gal	3·788 l
1 lb	0·453 kg
1 ton	1·016 tonne

1 ft/sec	30·48 cm/sec
1 m.p.h.	1·609 km/hr
1 lb/ft²	4·8819 kg/m²
1 lb/in²	0·0703 kg/cm²
1 atm	1·0334 kg/cm²
1 ton/in²	57·488 kg/cm²
Temp. °F	1·8°C + 32
Temp. °K	°C abs
1 B.Th.U.	0·2519 kg. cal
1 Therm	25198·0 kg. cal
1 ft-lb	0·1382 kg-m
1 h.p.	1·0138 cheval vapeur
Equiv.h.p.	$\frac{\text{S.T.(lb)} \times \text{V(m.p.h.)} \times 100}{375 \times \text{Airscrew eff. (per cent)}}$
Equiv.h.p. (nominal)	$\frac{\text{Static thrust (lb)}}{2{\cdot}6}$

Preface to the Sixth Edition

THE first, diminutive edition of this book, based upon a series of articles which appeared in *Flight* during the summer of 1941, was coaxed into print during the darkest of the war years. There was everything to discourage its appearance—air raids, paper shortage, security regulations, fire-watching and extra work to do on behalf of colleagues in the Services. Yet Geoffrey Smith wrote and compiled long into the noisy nights with a faith and enthusiasm characteristic of him, though in this instance transmitted in part from Frank Whittle and a handful of his co-workers. Eventually the book appeared, to throw early light on the gas turbine and its potentialities. At that time jet propulsion was pioneering stuff and near madness; it brought praise from some but ridicule and incredulity from others who should have known better. Germany, unknown to the British technicians, was working along the same path, while in America and Russia the first information was just being received of the portentous British development.

In his introduction to the American printing of the fourth edition the author recorded that early in 1941 he discussed the subject of propulsion by thermal jet reaction with Lord Brabazon, then Minister of Aircraft Production, pointing out that no book on the subject existed. Lord Brabazon readily agreed as to the desirability of reviewing designs, some bordering upon the practical stage, in several countries. He expressed the view that it would be a useful service to British engineering if the progress toward this new form of prime mover was discussed and interest centred in further research; but, of course, nothing at that time was to be revealed of British activities. Not until the official announcement on January 7th, 1944, that jet-propelled fighter aircraft had been adopted by the R.A.F. could credit be given to Frank Whittle and to this, his country.

In a preface to this same edition, General Carl Spaatz, as Commanding General of the U.S. Army Air Force, wrote: ‘The publication of an American edition of this British treatise is reminiscent of our collaboration for victory during World War Two. Present progress of jet propulsion for aircraft in the Army Air Forces is a direct result of the exchange of information and ideas between British and American engineers and scientists. Our first jet-propelled aircraft, the XP-59, might never have been built if a joint co-operative programme with the British had not been realized. The exchange of research data enabled an American manufacturer to produce the two Whittle-type power units which were used in the first flight of the XP-59.’

That progress was rapid as soon as the significance of this new prime mover had been grasped is confirmed by the size and content of this sixth edition.

In the preparation of each successive impression and edition the author, until his death in 1951, enjoyed the valuable assistance of F. C. Sheffield, who has now assumed the whole task of revision.

THE EDITOR,
Flight.

Dorset House,
Stamford Street,
London, S.E.1.

CHAPTER 1

Propulsion by Jet Reaction

BY A JOINT ANNOUNCEMENT in January 1944, Britain and the U.S.A. disclosed to the world the existence of jet-propelled fighter aircraft. Public imagination was immediately captured. Since that time the spectacular performance of advanced types of turbine-powered aircraft has served to maintain an interest which transcends those aroused by other major technological developments of the Second World War period. Gas turbines and jet propulsion have become accepted by the practical world whilst radar, working literally in the dark, and atomic fission, still shrouded in secrecy and associated with awe-inspiring powers of destruction, are less well understood and tend to be regarded as belonging solely to the domain of the scientist.

Jet propulsion had long been the subject of investigation in various countries but practical development was accelerated in the years before the War. In Britain, where much of the pioneering work was done, the secret of its success was preserved for several years, although full information was communicated in July 1941 to the appropriate authorities in the U.S.A. under the then existing Mutual Aid Agreement. British fighters of the new type, powered by two turbojet units, were first used operationally against the German V-1 flying bombs in the summer of 1944.

People who, on first introduction to the possibilities and advantages of flight without the aid of an airscrew, were apt to dismiss the notion as ' fantastic ' or ' futuristic ' may well be excused. When a series of articles was published in *Flight* in 1941, reviewing known designs and outlining the possibilities, many engineers and technicians were frankly sceptical of the practicability of jet propulsion. Owing to their training and long familiarity with the internal combustion reciprocating engine which had been developed to a high stage of efficiency, they were somewhat slow and seemingly reluctant to appreciate the enormous potential advantages of the new system of propulsion. The remarkable and well-established achievements of the orthodox airscrew-reciprocating engine combination had tended to obscure two facts:

(1) Self-propelled bodies in a fluid medium, aircraft in air and ships in water, progress by reaction to the displacement of a mass of the medium rearwardly. Basically, therefore, all aircraft are jet-propelled.

(2) Despite intensive development, the reciprocating engine with intermittent power impulses is fundamentally a retrogression

from rotary units with continuous production of power, such as the early wind and water wheels.

Furthermore, the recorded experience with thermal ducts for aircraft and with the early combustion gas turbines for producing mechanical power was unfavourable. The views persisted that metals could not be evolved to withstand continuously temperatures of 550° C to 850° C, that the individual efficiencies of the turbine and the air compressor were low, and that in combination the high ratio of negative to positive work (compression to expansion) rendered it impracticable.

It needed the inspiration of a young officer in the R.A.F., Frank Whittle, to realize that when employed at high altitude to produce a propulsive jet for high speed aircraft the lightweight combustion gas turbine became a practical possibility. This, indeed, was one of the rare successful cases of ' killing two birds with one stone.'

Flight by means of a power-driven, heavier-than-air craft is a relatively recent achievement in human history It was on December 17, 1903, that the Wright brothers in the U.S.A. first demonstrated the possibility by four short flights. The first was at a height of 10 ft and lasted only 12 sec while the final attempt that day achieved a duration of 59 sec and a distance of 284 yd. Allowing for the prevailing head wind that was equivalent to a ground speed of about 10 m.p.h.

By 1939 international records had been established for maximum speed at 464 m.p.h., for altitude at 56,000 ft and for non-stop distance to more than 7,000 miles. All this was accomplished by the aid of internal combustion engines of the reciprocating type. The Wright brothers' four-cylinder engine developed only 12 b.h.p. and weighed 240 lb or 20 lb/b.h.p. Just prior to the Second World War engines having power outputs in excess of 1,000 b.h.p. were produced at a specific weight of about 1.25 lb/b.h.p. Special racing engines, including those for the Schneider Trophy winner, scaled less than 1.0 lb/b.h.p.

In view of this phenomenal progress it may be wondered why, in a number of different countries, serious attention was devoted to the development of a new type of power unit and a new method of propulsion. That both the piston engine and the airscrew were capable of further improvement has since been amply demonstrated by the production and operation of engines of more than 3,000 b.h.p. and four- and five-bladed, variable pitch airscrews, and also counter-rotating airscrews, to absorb such tremendous outputs. Even this does not necessarily represent the ultimate.

The idea of jet propulsion was not novel. In fact, the fundamental ' equal and opposite ' effect of force and reaction was known to the ancients and had been embodied in a physical law by Isaac Newton more than 250 years earlier. Vessels had been propelled by rearwardly directed jets of water, and air jets had been tried for aircraft propulsion by the French military authorities during the First World War.

A little reflection, however, will show that the development was undertaken for the reason that most worth-while technical advances are made. It was in an attempt to meet a need which could not adequately be fulfilled by existing techniques. Aircraft builders striving for better performance at higher speeds and higher altitudes were handicapped in their efforts by the lack of high-powered engines of low specific weight and small bulk. As operational altitudes were raised they were confronted with the combined effect of a serious falling off of power output and reduced airscrew efficiency. When the latest and best reciprocating engines were offered, units developing more than 2,000 b.h.p. with a specific fuel consumption of 0.5 lb/b.h.p./hr and weighing only approximately 1 lb/b.h.p., the constructors demanded still more powerful units of less weight and reduced frontal area. In short, it was the problem, familiar to power engineers, of not merely forcing a quart into a pint pot, but of then extracting a gallon from it.

How was this demand to be met? The development of existing multi-cylinder reciprocating engines could be continued or a diversion could be made to seek an entirely different type of prime mover. The first method, however, had become excessively complicated and limited in its results. The alternative course, which seemed at first to present such hazards and difficulties, was found to be more easy and to offer greater potential advantages. Turbojet units had proved far simpler to design and quicker to produce than the orthodox engine. From the commencement of work in the drawing office to successful bench tests in about six months had already been achieved. Even under the impetus of war it required from four to five years to bring a large piston engine to the fully operational stage.

It must be conceded, however, that development work on gas turbines has tended to occupy more time as the units become more complicated in the search for fuel economy and long service. Painstaking investigations are a long term process and the solution of problems is not determinable at any precise moment.

After the War it seemed that mechanical improvements to reciprocating engines could offer gains of only a few per cent and no major changes were visualized. Radical improvement in fuel could well yield greater output, but would call for higher working pressures which would probably lead to increased specific weight, as in the diesel engine. As a result of intensive development it became apparent that the gas turbine was destined to challenge and eventually supersede the reciprocating engine for units of high power output.

The pre-war 'big' engine of 1,000 b.h.p. had already become a 'medium-power' engine, and units of approximately 3,000 b.h.p. were not uncommon. These large engines had 24, 28 and even 36 cylinders. Estimates of requirements in the immediate future suggested that engines of 5,000 b.h.p. to 10,000 b.h.p. would be demanded by aircraft constructors for the large high-speed aircraft then envisaged. Multi-bank, radial, flat, X-, and H-type units in this class had, in fact, been seriously investigated by engine-manufacturers in several countries. For an output of this order it had become a necessary practice to couple two or more standardized engines. Coupled twin-row radial engines were fitted to the mammoth Bristol ' Brabazon ' liner and in France 24-cylinder, H-type units had been operated in pairs. There was even a proposal to build four of these units into a composite plant making a total of 96 cylinders.

To increase power, therefore, the tendency is to employ a larger number of cylinders, which means that the total energy is applied to the driving shaft in a series of small impulses occurring in rapid succession. A six-cylinder engine has three power impulses per revolution or, in other terms, the firing interval is 120° of crankshaft rotation. On a 24-cylinder unit of the same capacity the firing interval would be 30° of crankshaft rotation, and there would thus be 12 smaller power impulses per revolution. When this feature is considered, it is obvious that the running gear (pistons, connecting rods and crankshaft) and also the general engine structure (cylinders, heads and crankcase) can be of relatively light construction. It will be seen that the greater the number of cylinders employed the nearer is the approach to a continuous application of energy. Of course, the number of cylinders cannot be increased indefinitely ; the cost, the complication and the number of component parts assume staggering proportions. When multiplying the number of cylinders ceases to offer practical advantage, another solution must be sought. Any substantial increase of the diameter of individual cylinders would involve structural and cooling difficulties. Similarly, an increase in the length of the piston stroke would raise inertia loads and, unless the rotational speed were reduced, would result in unfavourable piston speeds. Either would lead to an undesirable increase in the specific weight per h.p., thus rendering the engine less attractive for aircraft installation.

To improve engine performance it has become necessary to add or incorporate severally, multi-speed, multi-stage and exhaust turbine-driven superchargers ; fuel injection and water injection systems ; intercoolers and after-coolers ; ducted radiators and fan cooling ; variable-pitch, variable-speed and counter-rotating airscrews ; and even exhaust efflux propulsive systems. It will be noted that rotary and not reciprocating units are employed on both the compression and the expansion sides of the working cycle as superchargers and exhaust turbines. With so many variables and with such an aggregation of auxiliary equipment the long period required for design, development and test can be readily comprehended. The basic components, the cylinders and pistons, must perform four different functions in succession (induction, compression, expansion and exhaust), and in consequence each design is essentially a compromise and a successful unit cannot readily be scaled up or down. Some lessons may be learned and embodied in units of a different size but, in general, each engine must be designed as a whole.

A rotary power unit such as a gas turbine has its own peculiar problems but does not suffer the limitations of the reciprocating unit. Its chief merit is a relative simplicity in design, development and production which arises particularly owing to each component being called upon to perform only one function continuously. Within a considerable range it can be scaled up or down in size and output.

The working medium is air, which is compressed and then heated by the combustion of a hydrocarbon fuel. Partial expansion of the compressed and heated air through a turbine produces mechanical power which is used to drive the compressor, and continued expansion in the tail pipe produces a high-velocity jet. The reaction to this rearwardly directed jet furnishes the propulsive thrust. In the case of a turboprop unit expansion is carried further by more turbine stages, or by separate turbine, to produce further mechanical power to drive the airscrew. The relatively little energy remaining in the turbine efflux provides a supplementary propulsive thrust.

In operation a gas turbine has virtually no out-of-balance forces, and the consequent absence of vibration is calculated to extend the working life of the power unit and the aircraft and also reduce the strain on crew and passengers. It has no rubbing surfaces and the few bearings are usually of ball or roller type that can be given adequate lubrication without the losses involved in lubricating the sliding pistons of the orthodox engine. Weight and space occupied will be from one-third to at most one-half of those for multi-cylinder reciprocating engines of equivalent output.

British engineers, it is of interest to record, quite early began investigations into the possibilities of applying jet propulsion systems to aircraft. It will be an inspiration and encouragement to young engineers to know that Frank Whittle, to whom we owe the successful development of turbine jet propulsion for aircraft, was only 22 years of age when he lodged his first patent in 1930. For a time he experienced great difficulty in enlisting support for the development of his designs. Seven years later, by dint of painstaking experiments and after many disappointments, his first turbine-compressor unit ran successfully on the test bench.

METHODS OF JET PROPULSION

It is desirable that a differentiation be made of the various types of jet propulsion units and their respective methods of operation.

Power units may conveniently be grouped into four main classes:

(1) Rocket motors.
(2) Ramjets or athodyds.
(3) Pulsejets or intermittent-impulse ducts.
(4) Turbojets.

A number of subdivisions and variations can, of course, be made to these four main types. Although rightly included as a form of jet propulsion, the rocket differs essentially from the other thermal jet systems. Instead of carrying only fuel and drawing oxygen for combustion from the atmosphere, the rocket must carry both fuel and the necessary comburent. It thus furnishes the only means at present conceivable for space travel beyond the atmosphere surrounding the earth.

A rocket, however, is a true reaction unit and a brief examination of its functions serves to make clear the reaction principle on which all the thermal jet plants operate. If a hydrocarbon fuel and oxygen are held in a closed vessel and ignited (Fig. 1 : 1), the latent heat of the fuel is released, and there is a rapid expansion of the resulting gases. Owing to the fixed volume of the vessel, there will be a rise in pressure which will be uniformly distributed in every direction. As the forces are balanced there will be no tendency for the vessel to move.

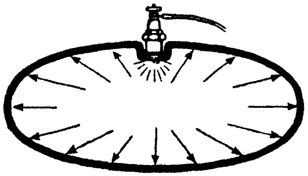


Fig. 1 : 1—Combustion in a closed vessel results in uniformly distributed internal pressure but no tendency for the vessel to move.

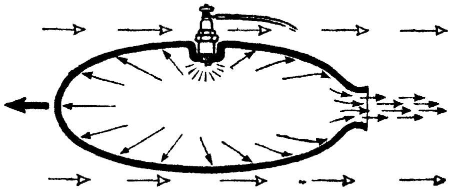
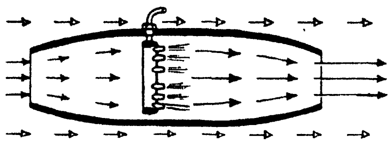


Fig. 1 : 2—Combustion in a rocket. The unbalanced internal pressure tends to propel the vessel in the opposite direction to the issuing jet.

Next consider a similar vessel which, instead of being sealed, has at one end an aperture or nozzle. On ignition and combustion of the fuel the expanding gases will rush out of the nozzle at a high velocity (Fig. 1 : 2). The internal pressure at the nozzle end of the vessel is thus relieved, leaving an unbalanced pressure at the other end which tends to propel the vessel in the opposite direction to the issuing jet. Obviously it is dependent solely upon internal conditions, and there is no suggestion whatever of the jet ‘ pushing against ’ surrounding external air. In fact it would function most efficiently in a complete vacuum.

Turning now to the thermal jet unit, we have a tubular duct with open ends attached longitudinally to an aircraft. When the aircraft is flying at high speed, air enters the forward end of the duct, is raised to

Fig. 1 : 3—The ramjet unit. Air rammed into a diffuser duct is heated and discharged through a convergent nozzle.



some pressure above atmospheric, is heated by the combustion of a liquid fuel and expands to leave the rear end as a high-velocity jet at a lower specific pressure. The amount of heat that can be added is largely dependent upon the pressure of the air being treated.

According to the Bernoulli theorem and assuming a steady, frictionless flow, the transport of energy by the air stream in the duct is constant. By suitable variation of the cross-sectional area of the duct, therefore, specific pressure can be increased at the expense of velocity or vice versa. For operation at subsonic air velocities the duct may be formed with a divergent entry nozzle, the effect of which is to build up the pressure of the air whilst decreasing its velocity.

A further advantage can be secured by forming the exit of the duct as a convergent nozzle. This has the opposite effect to the divergent entry nozzle and increases the velocity of the air stream whilst lowering its unit pressure. Fig. 1 : 3 shows a divergent-convergent duct into which fuel is injected and burned so that the heat is released directly to the air stream. A further slight advantage accrues as the products of combustion are entrained with the air flow and add to the mass of the propulsive jet. Despite the fact that it is incapable of providing initial power for take-off and has an inordinately high rate of fuel consumption, the principle is sound. Units of this type, known as ramjets or ' athodyds ' (an abbreviation of aero-thermo-dynamic-ducts), are being developed experimentally and, it is suggested, will have practical application for short-range interceptor fighters and controlled missiles. It is, of course, necessary to employ air launching or rocket-assisted take-off to attain the requisite speed to build up ram compression. Another possible use may be as blade-tip propulsion units for helicopter rotors, and experiments proceed along these lines.

A further type of jet unit (Fig. 1 : 4) also depends on ram effect produced by its forward motion and requires an assisted launch. The pulsejet, or intermittent-impulse duct, obtains an improved compression

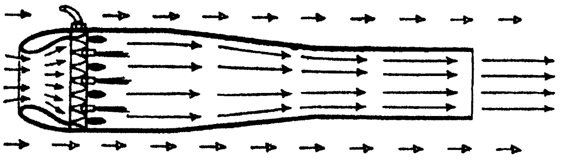


Fig. 1 : 4—The pulse-jet as employed on the German flying bomb.

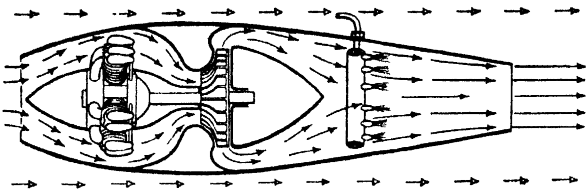


Fig. 1 : 5—Mechanical compression of the air by means of a conventional reciprocating engine, as in the Campini experiments in Italy.

of the air but only by the sacrifice of the principle of continuous power generation. It comprises a tube fitted at the forward end with a series of non-return admission valves and fuel injection nozzles. As it travels through the air, pressure on the nose opens the valves and air is rammed into the duct to mix with the fuel and form a combustible mixture. As the contents are heated by the combustion of the fuel, there is a rapid expansion and, by appropriate proportioning of the exit nozzle, a rise in pressure which serves to close the valves. The violent ejection of the gases creates a depression inside the duct which allows the flat spring valves to reopen and admit a fresh charge of air and the cycle is repeated. The frequency of the cycle will depend upon the designed proportions of the duct and may be relatively high. A propulsion unit of this type fitted to the German V-1 flying bomb had a frequency of about 2,800 cycles per minute.

In order to bring the jet propulsion unit to a stage where it had any prospects of competing with the orthodox reciprocating-type internal combustion engine it was necessary to mechanize the compression of the air. Reciprocating compressors have been suggested, but as the unit requires vast quantities of air and burns fuel continuously, the most appropriate component is a rotary compressor. Both centrifugal and axial-flow types are employed for this purpose.

This brings more complication in its train as, of course, mechanical energy is required to drive the compressor. An obvious means was available in the highly developed reciprocating-type aircraft engine. Fig. 1 : 5 shows a plant having an air-cooled radial engine driving a centrifugal compressor and subsequent heating of the air by a ring of fuel sprays. This is the principle of the Campini system.

The next step was to eliminate the anomalous reciprocating engine and substitute a simple rotary power unit. A combustion gas turbine was

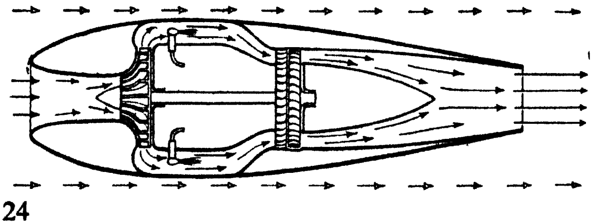


Fig. 1 : 6—Mechanical compression of the air by a gas turbine is the basis of the Whittle system.

mounted on a common shaft with the compressor and proportioned to extract merely sufficient energy from the issuing gas stream to furnish the necessary power to drive the compressor and ensure continuous operation. Fig. 1 : 6 shows diagrammatically this turbine-compressor combination, the Whittle system, which is in practical operation in the air.

It is not imperative for the turbine power unit to be allied with the system of jet propulsion, which is particularly suitable for aircraft operating at high speeds and high altitudes. In a similar manner to the gas turbines for railway, marine or stationary application it can deliver mechanical power and drive an airscrew. For medium-speed, medium-altitude craft, a turboprop unit can offer relative advantages with a lower specific rate of fuel consumption and better take-off characteristics than a jet unit.

In its simplest form the basic components, compressor and turbine, are mounted on a common shaft and the airscrew is driven from this shaft through a reduction gearing. The turbine is of the multi-stage type designed to expand the gas stream down to about atmospheric pressure. There are, however, a number of variant arrangements. Separate high-pressure and low-pressure turbines may be used to drive respectively the compressor and the airscrew, as in Fig. 1 : 7, or the h-p turbine may be arranged to drive a h-p compressor and the l-p turbine a l-p compressor and the airscrew.

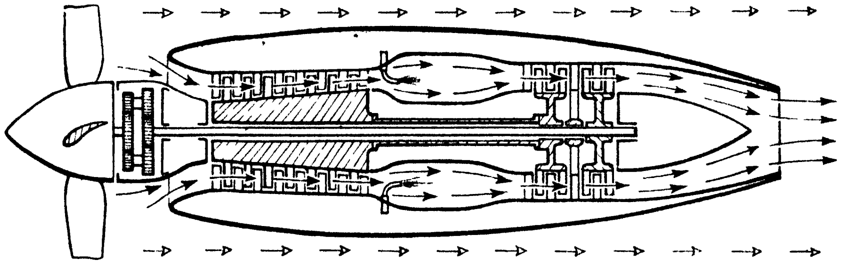


Fig. 1 : 7– Turboprop unit of the two-shaft type.

A COMPARISON WITH PISTON ENGINES

The reader is probably acquainted with the orthodox four-stroke reciprocating engine used in aircraft and motor vehicles, and with its cycle of operations—induction, compression, expansion and exhaust. However, for those who are not fully familiar with the cycle, a brief description is included. All four of the above functions take place in regular sequence in each cylinder of the engine. As the piston descends a mixture of air and petrol enters the cylinder past the open inlet valve. The inlet valve closes as the piston ascends to compress the mixture ready for ignition by an electric spark. The expansion resulting from combustion forces the piston downwards, and this is the sole power stroke. Finally the piston once more ascends to expel the burnt gases by way of the exhaust valve. In an engine operating at a crankshaft speed of 3,000 r.p.m. each stroke is completed in 0·01 sec and a complete cycle of operations in 0·04 sec. Owing to the reciprocating motion the piston must be accelerated, decelerated and its direction of motion reversed 6,000 times per minute. As the power stroke occurs only once in every two revolutions of the crankshaft, the working energy is applied to the shaft intermittently. The firing frequency in a particular engine depends, of course, upon the number of cylinders.

Fig. 1 : 8 is a series of sketches showing a complete cycle of four strokes and immediately above is a sectional diagram of a two-shaft gas turbine, as in a turboprop unit. The positions of the two diagrams have been arranged in order to show their corresponding functions in relation. In comparing the action of the two types of engines it is important to

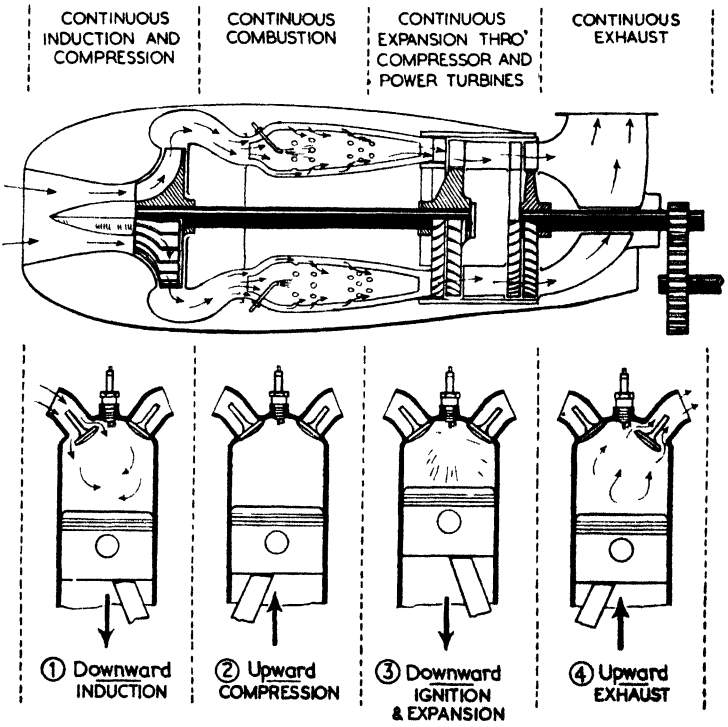

CONTINUOUS INDUCTION AND COMPRESSION
CONTINUOUS COMBUSTION
CONTINUOUS EXPANSION THRO' COMPRESSOR AND POWER TURBINES
CONTINUOUS EXHAUST
① Downward INDUCTION
② Upward COMPRESSION
③ Downward IGNITION & EXPANSION
④ Upward EXHAUST

Fig. 1 : 8—A comparison of rotating and reciprocating power units. In the lower series of diagrams the four intermittent strokes of an orthodox piston engine are approximately aligned with the four continuous functions of a two-shaft turbine.

appreciate that the turbine produces power continuously; that is, there are no intermittent explosions as in a reciprocating engine.

HOW THE GAS TURBINE FUNCTIONS

A gas turbine unit consists of three essential components, (1) a rotary air compressor, (2) one or more combustion chambers into which a liquid fuel is sprayed and burned continuously, and (3) a turbine revolving with the compressor on a common shaft. There are no pistons, valves or ignition apparatus, save for two igniter plugs to attain a start. Initially an electric starter motor spins the main shaft at about one-eighth operational speed. The starting cycle is automatically controlled by a time switch. The pilot opens a fuel cock, depresses a button, and in about half a minute the unit is running at idling speed. Virtually no ' warming-up ' period is necessary as bearings are few, there are no metal-to-metal sliding surfaces, and consequently only a simple lubricating system.

At the start, air is induced at the front by the action of the rotating compressor, forced by the same compressor into the combustion chambers at a ratio of about 4 : 1 and into this compressed air fuel is pumped at a pressure of about 750 lb/in^2 and burned continuously, somewhat in the manner of a plumber's blowlamp. In flight the work of compressing the air is greatly assisted by the forward motion of the aircraft. Air is rammed into the intake duct and at speeds over 300 m.p.h. the effect is of such magnitude that the thrust curve ceases to fall and commences to rise. The lower air temperatures encountered at high altitudes also improve the efficiency of the compression system.

Only a small quantity of air is used for combustion to provide the necessary air : fuel ratio of about 14 : 1. The major portion—the so-called ' diluent ' air in distinction to the ' combustion ' air—lowers the air : fuel ratio to about 60 : 1 and reduces the combustion temperature of, say, 1,800° C to about 850° C at the entry to the turbine. The heat released by the combustion of the fuel expands the air but as the chambers have an open end, in contrast to the closed cylinder of a piston engine, there is no rise in pressure. Actually there is a slight fall in pressure, about 2 or 3 lb/in^2, from the entry to the outlet of the chambers.

By reason of the rapid expansion, the heated air and combustion products at increased velocity force their way through the only exit from the chambers, namely, between the guide vanes in the stator ring of the turbine. These vanes direct the flow to the appropriate angle of attack for the blades on the periphery of the turbine wheel. The flow of the heated mixture past the aerofoil-shaped blades causes the wheel to rotate rapidly. Maximum speeds of 8,000 to 16,000 r.p.m. are typical for main propulsion units but small auxiliary power turbines may run at twice these speeds. In this manner mechanical energy is created at the shaft. The compressor mounted on the front end of the shaft is rotated to com-

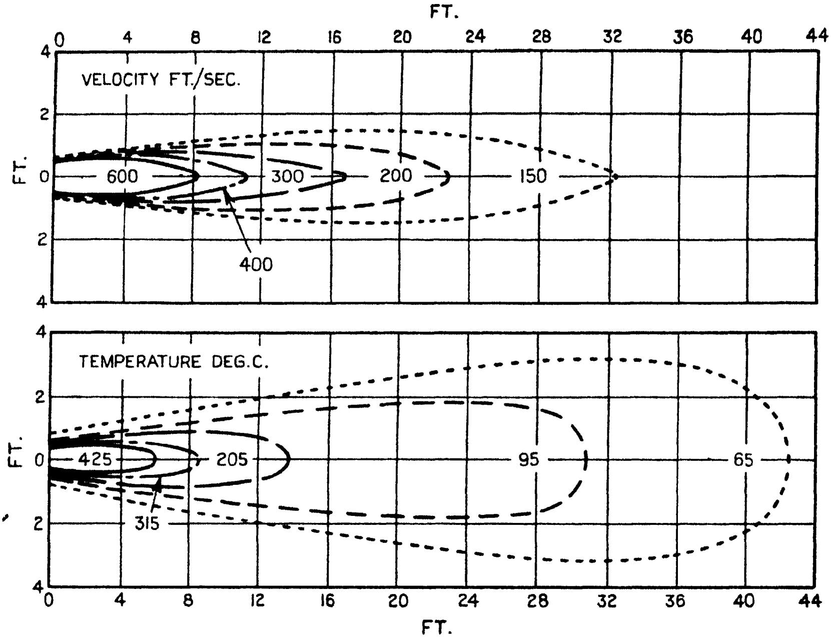


Fig. 1 : 9—Velocity and temperature patterns in jet wake.

press more air into the combustion chambers, and the process continues so long as fuel is introduced and burned. As in the reciprocating engine, less fuel is required as altitude is increased and a barometric device is fitted to regulate automatically the supply injected into the combustion chambers. The usual fuel is kerosine but petrol or diesel oil can be employed. Calorific value is the prime fuel characteristic and the specific calorific value per gallon is of more importance than the value per unit weight as in high-speed aircraft weight rather than bulk can be the more readily accommodated. //

In a pure jet unit all the power delivered by the turbine is expended in driving the air compressor and the necessary auxiliary equipment. To drive the compressor approximately 100 b.h.p. is required for each lb of air delivered per second. A pound of air per second gives about 50 lb thrust, therefore the turbine of a unit developing 2,000 lb thrust must deliver about 4,000 b.h.p. From the rear of the turbine the heated air, still expanding, issues from the tail nozzle as a jet, having a velocity that may exceed 1,200 m.p.h. or 1,760 ft/sec. The temperature at the outlet is down to about 600° C, compared with 690° C as it enters the jet pipe. Specific pressure is relatively low, being only from one-and-a-half to twice that of atmosphere. In Fig. 1 : 9 are shown velocity and temperature distribution patterns in the wake of a small jet unit developing a thrust of about 600 lb.

A summarized list of the potential advantages of turbine propulsion units as compared with orthodox reciprocating engine installations follows :

(1) They can be designed, built and tested in considerably less time than is usually required for piston engines.
(2) Production is simpler and speedier. They have only about one-third the number of parts required for comparable piston engines.
(3) As turbine components each perform one specific function, a proved unit can be more readily scaled up or down to meet power requirements.
(4) They can be designed and built for much higher individual power outputs. Practical considerations adversely affecting the power : weight ratio limit the output of piston engines.
(5) They are less critical regarding fuel. High calorific value, not anti-knock rating, is the criterion. Kerosine is the usual fuel but petrol or diesel-oil may be employed.
(6) Power is produced continuously instead of intermittently. Consequently working pressures are low and the structure, casing and ducting are of light section and weight.
(7) By the exclusive use of rotating components, vibration is virtually eliminated. Weight can, therefore, be saved on the airframe.
(8) Absence of reciprocating parts enables higher operating speeds to be employed. Frontal area and space occupied are reduced and power : weight ratio is improved. Turbojets are about one-third the weight of comparable piston engines.
(9) As there is no sliding friction (pistons in cylinders) and no heavily loaded bearings, lubrication is greatly simplified.
(10) They are internally self-cooled by the air flow and require no radiators or projecting air scoops.
(11) They are easy to install and remove from the airframe and simple to service. Some units can be removed in less than half an hour.
(12) Auxiliary components, instruments and controls are all reduced in number. Less attention is demanded from the pilot.
(13) They require no preliminary warming-up and will develop full power for take-off in two or three minutes.
(14) They operate more efficiently at high altitude and consequent low temperature than at sea level. They need no complicated supercharging system to maintain power at altitude.
(15) The higher the aircraft speed the more efficiently they function due to the influence of ram pressure.
(16) Size, shape and weight of a turbine unit facilitate complete enclosure in the fuselage or wing.
(17) By suitable proportioning of the components the main compressor system can be arranged to supply air to charge a pressure cabin, to warm armament enclosures or to prevent icing of control surfaces.
(18) The absence of an airscrew on turbojet units permits the use of a relatively light and low undercarriage. On flying boats the absence of an airscrew reduces one hazard.
(19) For military aircraft, turbojet units permit better location, an unrestricted field of vision for the pilot and unobstructed forward armament.
(20) Vibrationless running, absence of mechanical noise and reduced exhaust noise make turbine units particularly desirable for civil aircraft.

CHAPTER 2

Early Projects

According to an old saying 'there is nothing new under the sun,' and certainly jet reaction, jet propulsion and the turbine may be cited as some of the earliest mechanical ideas. Students may recall the æolipile demonstrated by Hero, the Alexandrian philosopher, at about the commencement of the Christian era. This apparatus consisted of a hollow sphere mounted to rotate between two pillars, one of which, being hollow, served to transmit steam from a closed vessel supported over a fire. Two pipes with right-angled ejection nozzles were mounted diametrically opposite on the sphere in a plane normal to the axis of rotation and the reaction to the issuing jets of steam caused the sphere to revolve.

Sir Isaac Newton, the Lincolnshire scientist, as long ago as 1687 had recognized the possibilities of jet propulsion by the formulation of his third law of motion that action and reaction are equal in magnitude and opposite in direction, but he did not originate the experiment illustrated by Fig. 2 : 2. Newton's philosophy in 'Principia' was the subject of a later work by Gravesande entitled 'Natural Philosophy,' which had for its sub-title 'An Introduction to Sir Isaac Newton's Philosophy.' A contrivance of the nature depicted was therein mentioned as an illustration of Newton's law, but not as having been devised by him. As a result, many text-books which followed included a picture of a carriage propelled by jet reaction and loosely captioned 'Newton's steam carriage.'

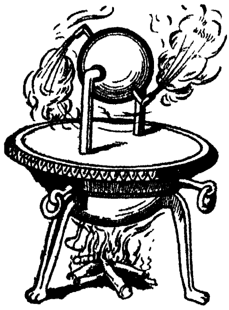


Fig. 2 : 1—Hero's æolipile is reputed to be the first apparatus converting steam pressure into mechanical power. It was probably the earliest demonstration of the principle of jet reaction.

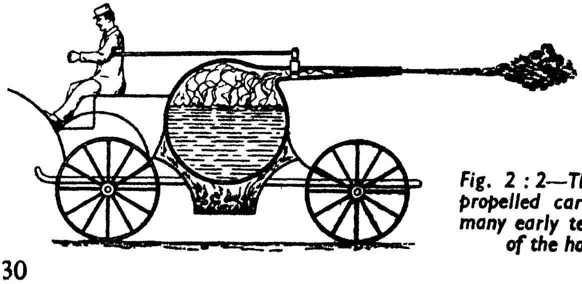


Fig. 2 : 2—The illustration of a jet-propelled carriage which appears in many early text-books on the evolution of the horseless road vehicle.

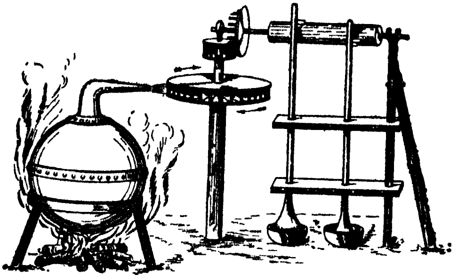


Fig. 2 : 3—Possibly the first proposal for a turbine was made by an Italian engineer, Giovanni Branca. The value of high rotational speed and reduction gearing was apparently realized in 1629.

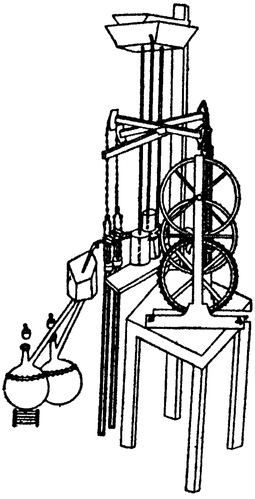


Fig. 2 : 4—The first gas turbine proposed and patented by John Barber in 1791. (Patent No. 1833.)

It showed a four-wheeled vehicle with a spherical boiler, mounted over a fire, the top of the boiler having a rearwardly directed nozzle. By the reaction to the jet of steam thus created the carriage was to be propelled in a forward direction. Speed was to be regulated by the driver's control of a steam cock in the issuing pipe.

In the British Museum is a model of a machine ascribed to Giovanni Branca, an Italian engineer, who in 1629 suggested a form of steam turbine. His idea was to direct a powerful jet of steam tangentially against open vanes formed on the circumference of a fan wheel mounted on a vertical spindle. From this spindle other mechanisms were to be driven at reduced speed through a crude cogwheel gearing. Fig. 2 : 3 shows the turbine arranged to drive a stamp mill.

In 1791, that is over 160 years ago, an Englishman, named John Barber, took out the first patent for a gas turbine. The sketch which accompanied it was of a very rudimentary character and certainly it provided no clue to the technical possibilities of the design. The proposed plant comprised a gas producer, gas receiver, gas and air compressors, a combustion chamber, a turbine wheel and speed-reducing gearing. About that time the substitution of mechanical power for human energy had become an economic necessity and, apart from wind and water wheels, the steam engine was the only known machine capable of producing motive power. There is no information as to what influenced him to devise an alternative to the ubiquitous steam engine. Relative efficiency would not arise at that early period, and it can only be assumed that it was desired to produce a prime mover less complicated than the boiler,

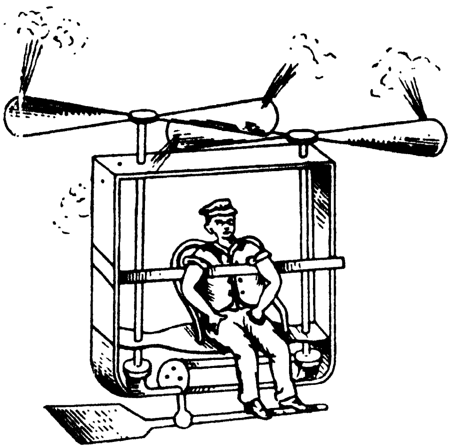


Fig. 2 : 5—Reproduced from an 1860 sketch of a steam-jet reaction-propelled helicopter proposal.

engine and condensing equipment of the steam plant and to obtain rotary motion without the need for the crank and connecting rod mechanism.

A concluding example of inventive vision is the ‘flying machine’ embodying the principle of Avery's rotary steam engine. The illustration is from *Flying*, an Iliffe periodical of 1903. Airscrews and helicopter rotors driven by blade-tip jet reaction devices are currently being investigated in several countries.

At the beginning of the present century the practical realization of the age-old dream of mechanical flight was delayed by lack of an efficient and reliable power unit. Steam plants were too heavy and cumbersome and the internal combustion engine was in its infancy. With little design or operational experience to draw upon, with quite elementary production facilities and with unsuitable and untested materials, the pioneers were obliged to build their own engines as well as the airframes. They developed their technique as they experimented and suffered many bitter disappointments. It was natural that in looking ahead such men sought alternative means of power production and methods of propulsion.

For a brief period the Gnome engine, that ingeniously conceived, beautifully made, but temperamental combination of rotary and reciprocating motion, met current requirements and made possible tremendous progress in aeronautics. However, since the advent of practical mechanical flight the demand has been for more and still more power from smaller and lighter units. It is of more than passing interest to record that jet propulsion, gas turbines, ramjets, thrust augmenters and thrust spoilers were visualized by the pioneers. Moreover, it is only as the result of successful development of the gas turbine and jet propulsion that engine manufacturers were able, for the first time, to supply power units that met fully the requirements of the airframe builders. For a time, power output no longer exerted a brake on progress. Currently, the demands for civil aircraft can still be met but again the builders of military aircraft are calling for higher powers.

It was probably inevitable that the earliest efforts should be made in conjunction with the orthodox internal combustion engine. Perhaps the simplest example is provided by the 1908 scheme (Fig. 2 : 6) of the French engineer, Lorin. In this a conventional engine exhausts directly through a divergent nozzle to produce a propulsive jet. Air is compressed in the cylinder, there is combustion of fuel and the effluent expands in the discharge duct. Apart from the energy used in driving the usual auxiliaries, the engine functions solely to produce the jet and no power is taken off the engine crankshaft. Lorin visualized multi-cylinder units of this type, as shown in perspective, installed in the wings of aircraft.

By modern standards this design is only of historical interest. Nevertheless it deserves honourable mention, as it was the original suggestion to employ a reciprocating engine and use part of the combustion energy to compress the air. It suffered, of course, by reason of the quite inadequate mass of air handled, which could not be more than that required to produce a combustible mixture in the engine cylinder.

Early efforts to overcome this handicap led to ingenious schemes to increase the mass of air and thus ' augment ' the thrust. In 1917 O. Morize, of Chateaudun, France, proposed the plant shown diagrammatically in Fig. 2 : 7. An engine A drives a compressor B which delivers air through an equalizing chamber C to the combustion chamber D, the walls of which are lined with refractory material. The air inlet surrounds the jet E supplied with liquid fuel from tank F by an engine-driven pump G. A suitable means of ignition, electric or incandescent, is provided, and the combustion products are discharged through the combustion

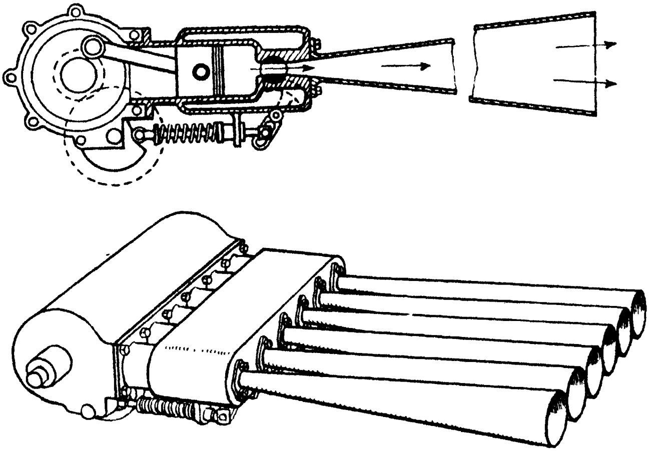


Fig. 2 : 6—The Lorin scheme of 1908. A conventional engine is employed solely to produce a propulsive jet. The lower sketch shows a proposed multi-cylinder layout for wing installation.

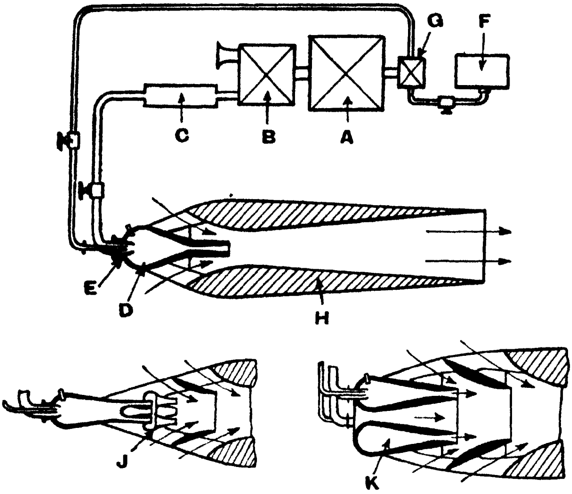


Fig. 2 : 7—In the Morize ' ejector ' scheme, fuel and air are supplied to a combustion chamber which discharges into a convergent - divergent tube.

A. Reciprocating engine.
B. Air compressor.
C. Equalizing chamber.
D. Combustion chamber.
E. Fuel injector.
F. Fuel tank.
G. Fuel pump.
H. " Ejector " tube.
J. Multi-nozzle combustion chamber.
K. Annular combustion chamber.

chamber nozzle into the convergent forward end of the ' ejector ' tube H. This creates a region of low pressure and air is induced into the end of the tube H to join the rapidly moving gas stream. In the divergent portion of the tube H the velocity is diminished, thus imparting increased pressure to the stream as it is discharged.

Control of the thrust is effected by varying the quantity of fuel and the pressure of the charging air which supports combustion. Modified types of ' ejectors,' of which two are shown, had a series of air intakes, a plurality of combustion chamber nozzles, J, or an annular combustion chamber K.

In the same year H. S. Harris, of Esher, devised a propulsion plant of considerable interest, although it was not developed to a practical stage. He also sought to induce air into the discharge tube in order to increase the mass of the jet. Instead, however, of achieving this aim by allowing the combustion gases to expand in a tube of increasing cross-sectional area, he inverted the process adopted by Morize. The combustion gases enter the discharge tube at high velocity and low pressure (approximately atmospheric pressure) in the form of a cylindrical stream or jet. As this jet is cooled and contracted on its way to the discharge nozzle a ' depression ' is produced which induces the inflow of air at the open forward end of the discharge tube. The tube is of a considerable length in order to ensure that the stream of gas and air issuing from the discharge nozzle is completely cooled. Consequently, no rapid expansion will occur at the nozzle which might give a pressure drop and induce a reverse flow. To compensate for the continued cooling and the corresponding decrease in volume of the stream, the discharge tube is of decreasing cross-sectional area, thus maintaining the velocity.

It is somewhat remarkable that the scheme contains no suggestion of utilizing the effluents of the engine which drives the blower supplying air to the combustion chambers. On the other hand, it appears ahead of its time in the proposal to employ powdered coal for fuel, as an alternative to liquid hydrocarbons such as petrol or paraffin. It is noteworthy as being one of the earliest to employ a centrifugal compressor.

Petrol is used for the twin-jet plant shown in Fig. 2 : 8. A twin-cylinder engine A drives directly a low-pressure blower B, which draws air from a forward axial intake and delivers to combustion chambers C. Fuel from tank D is supplied to jets E and mingles with the air to form a combustible mixture. Each combustion chamber is positioned axially in a long discharge tube F open at both ends. The issuing gases, on cooling in this tube, produce a partial vacuum which draws air into the tube in the direction of flow and increases the mass of the rearwardly flowing stream.

A number of modified arrangements were suggested. Instead of two jets, a single large discharge tube with a plurality of combustion chambers may be provided. Or a series of combustion chambers may be fitted in each discharge tube and additional air be admitted at several points along the tube at successive stages. The charging blower, it was suggested, could be driven by means of a rotor mounted in the discharge tube; in which case, presumably, the engine would be used only for starting. Either the complete discharge tube or a terminal portion may be mounted to swivel horizontally and vertically to effect or assist the control of the aircraft. Furthermore, means of temporarily closing the discharge end of the tube may be provided to cause the gas stream to issue from the forward end to produce a braking effect when landing.

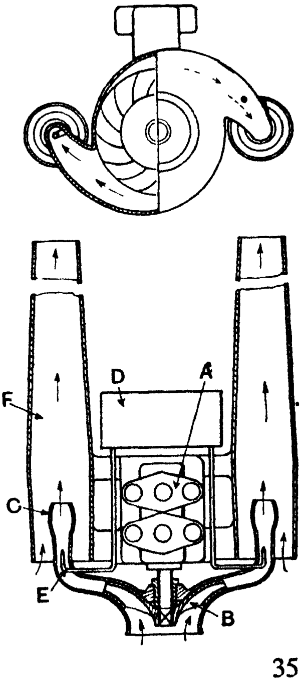
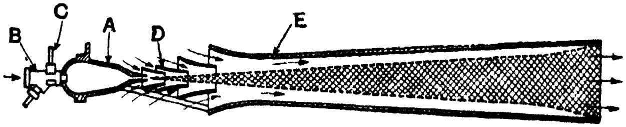


Fig. 2 : 8—The Harris plant also uses induced air, but the combustion chambers discharge into divergent-convergent tubes.

A. Twin-cylinder engine.
B. Low pressure blower.
C. Combustion chamber.
D. Fuel tank.
E. Fuel injector.
F Discharge conduit.

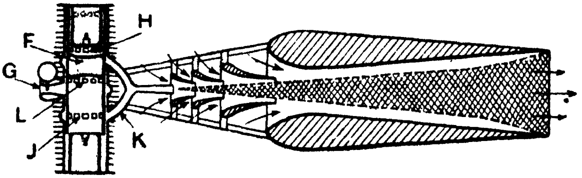
Many schemes have been proposed in which a stream of gases issuing from a combustion chamber creates an area of negative pressure in a tube or duct and induces a supply of air to augment the mass of the propulsive jet. One of the best-known and most widely quoted is that invented by Mélot and patented in 1920. Previously it had been submitted to the French military authorities who, during the First World War, experimented with the device. Referring to the diagram of the Mélot multiple-nozzle device (Fig. 2 : 9) the combustion chamber A is charged by relative wind entering by a forwardly facing venturi B and from a jet C, carrying in fuel which is ignited by electric or other means. From the chamber the combustion gases are emitted as a stream through a series of nozzles D of increasing diameter through which additional air is induced into the main diffuser tube E as indicated. The propulsive thrust was certainly 'augmented' by the additional mass of air, but not to the degree necessary to enable it to challenge the efficiency of the piston engine-airscrew combination.

One reason for the poor efficiency was the low compression of the fuel-air mixture in the combustion chamber. This was apparently appreciated by Mélot, for later he produced the relatively high-compression, combustion engine design, which is illustrated in Fig. 2 : 10. The air-cooled, two-stroke engine had two opposed cylinders in axial alignment, and a common, free-flying piston F having ported sleeve extensions. In the position shown a charge of air and fuel has been



A. Combustion chamber.
B. Venturi. C. Fuel injector.
D. Nozzle rings.
E. Main diffuser tube.

Fig. 2 : 9—A Mélot multiple-nozzle " thrust augmenter " of the type investigated by the French military authorities in the 1914–1918 war period.



F. Free-flying piston.
G. Venturi.
H. Piston upper sleeve inlet ports.
J. Piston lower sleeve exhaust ports.
K. Exhaust conduit.
L. Piston lower sleeve inlet ports.

Fig. 2 : 10—Later Mélot design in which a two-stroke engine is employed to furnish a higher compression cycle for the more efficient combustion of the fuel.

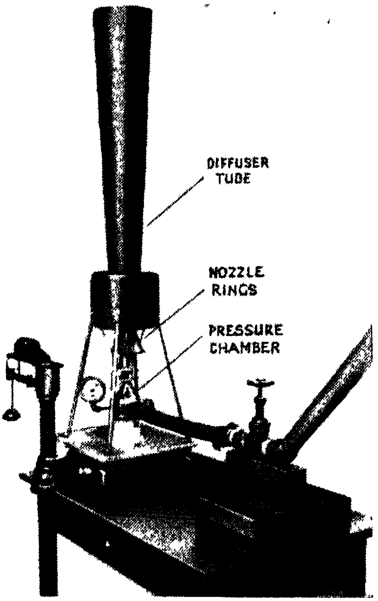


Fig. 2 : 11—Mélot type thrust augmenter on test at the Langley Memorial Aeronautical Laboratory, U.S.A.

admitted through the venturi G, the common cylinder inlet duct and the piston inlet ports H and compressed in the upper cylinder, which is at the point of ignition. The lower cylinder has discharged its effluents through piston exhaust ports J and the lower branch of the exhaust conduit K which delivers to the multiple-nozzle device. A new charge is simultaneously being aspirated by the lower cylinder by way of inlet ports L. Starting is effected by compressed air and electric ignition, but compression ignition is relied upon for continuous running.

In 1927 two American investigators, E. N. Jacobs and J. M. Shoemaker, undertook a series of tests at the Langley Memorial Aeronautical Laboratory to determine to what extent the Mélot type augmenter would increase the thrust reaction of a jet. The apparatus (Fig. 2 : 11) followed as closely as possible the Mélot design with a pressure chamber, three annular nozzle rings, and a main convergent-divergent diffuser tube. Instead of burning a fuel to raise pressure, however, the chamber was supplied with compressed air at normal temperature. The venturi-shaped nozzle of the pressure chamber had a throat of $\frac{3}{8}$ in diameter, and the main diffuser tube was about 3 ft 9 in in overall length. Mathematically it would appear that the thrust of a free jet would need to be increased several times in order to make jet propulsion a feasible proposition. Accordingly, the reaction thrusts of a free jet and a jet with Mélot and other type augmenters at various pressures were weighed on the scale.

The tests showed that the Mélot design gave the best results and represents a very creditable advance. Quantitatively, however, although the performance could possibly be improved by careful design, these results provided no evidence that the efficiency could be raised to the degree necessary for the successful propulsion of aircraft. The diffuser tube was designed to expand air from 185 lb/in^2 (gauge) to atmospheric pressure at the mouth, but the best results were obtained at approximately 100 lb/in^2 (gauge). At this pressure the thrust was 137 per cent of the theoretical thrust of a free jet, which was used as a basis for comparison, while a free jet was only 88 per cent. With a pressure of 185 lb/in^2 (gauge), the thrust of the Mélot augmenter fell to 122 per cent, while the free jet slightly improved to 90 per cent. This would seem to indicate that the diffuser employed was unsuitable in conjunction with the Mélot nozzzles and did not provide for adequate expansion of the jet from the pressure chamber. It was found that variation of the distance from the mouth of the pressure nozzle to the entrance of the diffuser between 2 in and 9 in produced no substantial effect on the thrust. A spacing of 6 in, approximately that used in the Mélot system, gave the best results.

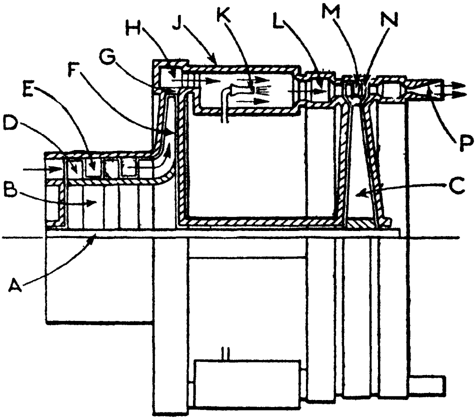
The French engineer René Leduc is mainly associated with projects for compressorless jet-propelled aircraft, so-called 'ramjets,' in which the main fuselage structure forms a continuous air duct from nose to tail. It is slightly barrelled to produce a divergent (compression) intake and a convergent (expansion) outlet. Due to the forward speed of the aircraft, air is rammed into the intake, the velocity is lowered and pressure increased by the divergent shape of the duct, liquid fuel is injected and burned and the combustion gas and air discharge from the exit nozzle at lowered pressure and increased velocity to form the propulsive jet. Such aircraft would need to be launched at high speed by some means as it would be quite impracticable to obtain sufficient thrust for take-off by air induced whilst on the ground and either stationary or moving at a relatively slow speed.

In 1938, at the Paris Salon de l'Aviation, a model of a Leduc ramjet aircraft was exhibited. A notice suggested that this 'machine of the future' would have an output of 14,000 h.p., a speed of 1,000 km/hr and a ceiling of 30 km. Nothing further was heard of this design until after the Second World War, when an experimental craft was built and flown.

On January 16, 1930, Frank Whittle filed the application for his first jet propulsion patent. It is a quite brief but extraordinarily interesting document which, in the cautious phraseology of the patent specification, reveals the inventor's aims, objects and opinions. It refers to a method of propulsion in one direction by the reaction caused by expelling fluid in the opposite direction, and is deemed to be particularly adapted for aircraft, but not necessarily limited to this use. The heat cycle is described as 'consisting of one or more stages of compression, one or more stages of expansion and a heat addition between the end of compression and the beginning of expansion, part of the work done in expansion being employed to do the work of compression, and the remainder to provide the fluid reaction.'

Characteristic of the inventor is the unequivocal expression of the belief 'that an embodiment of this invention will provide a large thrust in proportion to its weight, that it will perform at greater altitudes than are at present obtainable, that it makes possible higher speeds than have up to the present been obtained, that it will operate with any fuel now in use, and that it will have a reasonably low fuel consumption. Further, that simplicity and convenient external form is achieved.'

The propulsion unit proposed is shown diagrammatically in Fig. 2 : 12. On a common shaft A are mounted the rotor B of an axial-centrifugal air compressor and the gas turbine rotor C. The compressor rotor has axial flow blade rings D, working in conjunction with stator blade rings E on the casing, and terminal radial blading F. Air is delivered past diffuser vanes G to a hollow collector ring H and thence to a plurality of combustion chambers J. These chambers may be lagged to conserve heat and lined with refractory material, and liquid fuel is introduced and burned at jets K. From the combustion chambers the heated gas-air mixture passes into a collector ring L and expands through the turbine, which has a single stator blade ring M between two rotor blade rings N. After leaving the turbine the gases enter a third collector ring and are finally expanded to atmosphere through a plurality of axially disposed nozzles P.



A. Shaft.
B. Compressor rotor.
C. Turbine rotor.
D. Compressor rotor blades.
E. Compressor stator blades.
F Radial blades.
G. Diffuser vanes.
H. Air collecting ring.
J. Combustion chamber.
K. Fuel jet.
L. Gas collector ring.
M. Turbine stator blades.
N. Turbine rotor blades.
P. Discharge nozzle.

Fig. 2 : 12—The plant suggested by F. Whittle in his first patent specification, dated 1930, embodied a combined axial-centrifugal compressor and a plurality of combustion chambers.

It is stated in the specification that 'It can be demonstrated that the efficiency of this device conceived as a propulsive engine will not be reduced by reduction of the density of the atmosphere, and owing to the low temperature of the upper atmosphere may actually be enhanced.' Also, 'Controlling means may include fuel control, gas flow control, or mechanical control of the speed of the blower and/or its mover. The final emission of gas may perhaps be directionally controlled for manœuvring purposes.'

The first Whittle jet to be built, shown on the test bed in Fig. 2 : 13,

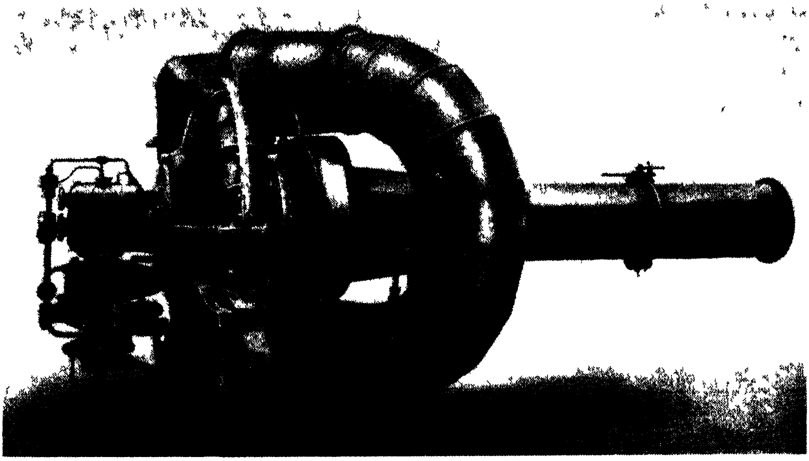
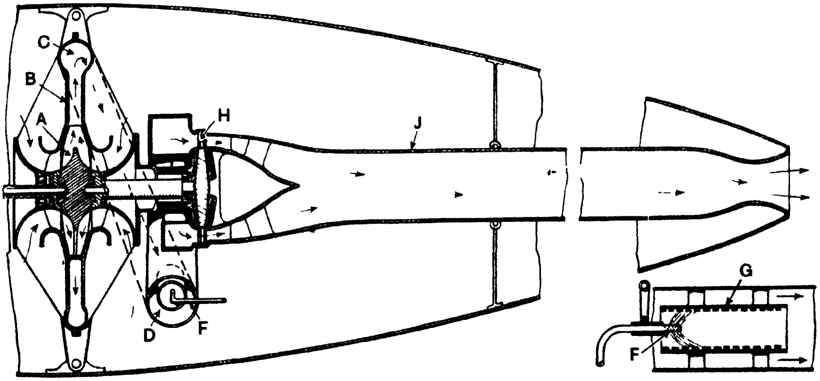


Fig 2 13—In 1937 the first Whittle jet unit was built. Here it is shown as assembled for static tests.



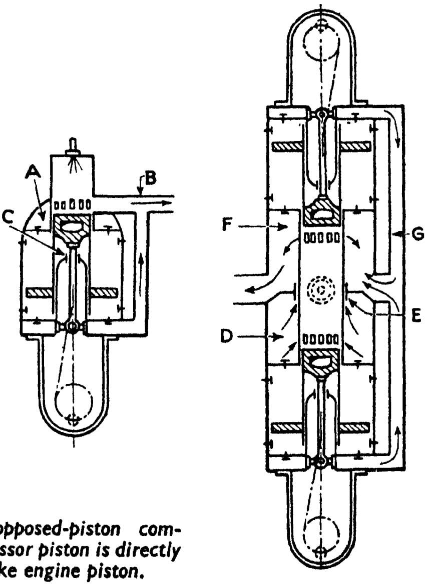
A. Compressor impeller.
B Primary diffuser chamber.
C. Air delivery scroll.
D. Helical combustion chamber.
F Fuel injector.
G. Combustion cowl.
H. Turbine rotor.
J. Discharge conduit.

Fig. 2 · 14—The first Whittle propulsion plant had a double-entry centrifugal compressor and a single combustion chamber. The turbine casing and rotor bearings were fluid-cooled.

had only one combustion chamber and the turbine casing was jacketed for water cooling. From the sectional drawing in Fig. 2 : 14, it will be noted that the centrifugal compressor was of the double-entry type. Leaving the periphery of the impeller A at supersonic velocity, air was delivered first into the radial primary diffuser chamber B, which discharged into the delivery scroll C, of increasing cross-sectional area. At its outlet the scroll was directly connected to a helical combustion chamber D. Initially the combustion chamber was of tapering form so as to constitute a secondary diffuser for the compressor, and the discharge end of the chamber was connected to the volute turbine nozzle. This extended around the periphery of the turbine, constituting a single annular orifice so that the blade ring of the turbine rotor was at all times open to the gas flow.

Fuel was introduced to the combustion chamber through injector F, surrounded by a cowl G. To ensure continuous combustion, the inner wall of this cowl could be covered with a perforated metal or a wire mesh (as indicated in the small detail drawing in Fig. 2 : 14) to produce a boundary layer of air travelling at a velocity below that of the flame speed of the fuel. The air and combustion gases gave up part of their energy to the turbine rotor H to drive the air compressor. After leaving the turbine, the air passed into an annular collecting chamber formed as a divergent channel between the discharge conduit J and a cone mounted on the turbine casing. It should be noted that in both the patent specification and in the first experimental unit to be built, a straight-through combustion system was employed although reverse-flow chambers were used on certain subsequent designs.

Turning to Switzerland, we find the system proposed by W. Schurter, of Zürich, makes exclusive use of compressor aggregates of the reciprocating type. Single-piston or opposed-piston, high-compression, two-stroke engines are employed, and both types are illustrated in Fig. 2 : 15. Dealing first with the smaller unit, the single-acting engine piston is rigidly combined with a double-acting, annular compressor piston. Groups of admission and discharge valves are provided at each end



A. Scavenging air trunk.
B. Combustion gas and air conduit.
C. Cushioning cylinder.
D. Scavenging and cooling air trunk.
E. Annular air passage.
F. Gas/air mixing chamber.
G. Upper and lower air conduits.

Fig. 2 : 15—Schurter single-piston and opposed-piston compressor aggregates. A double-acting compressor piston is directly attached to each single-acting, two-stroke engine piston.

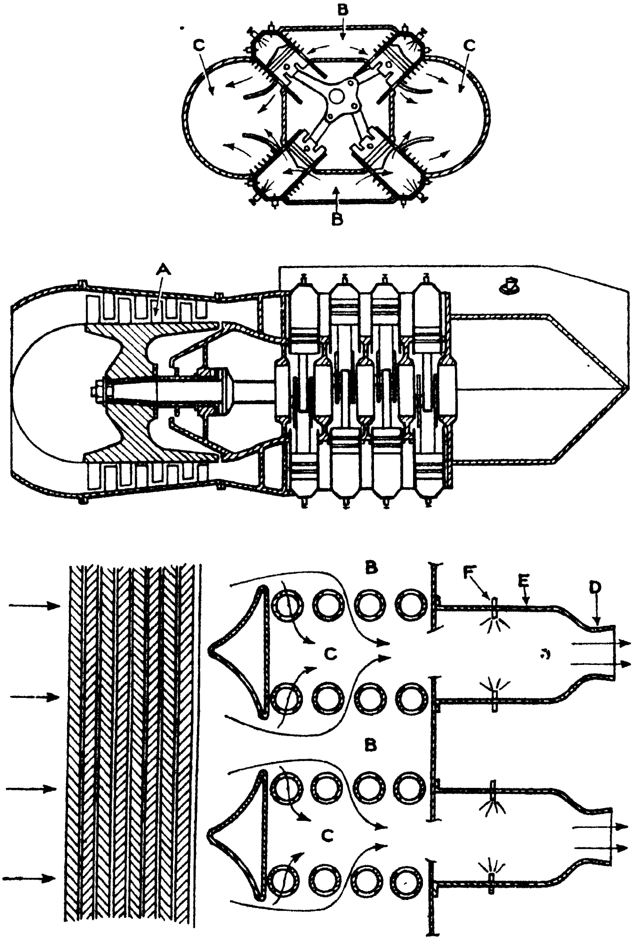
of the compressor cylinder. One end of the compressor delivers scavenging and charging air, at approximately twice atmospheric pressure, to the engine cylinder by way of duct A and the scavenge ports. From the other end air is discharged into conduit B, where it mixes with the combustion gases as they leave the engine cylinder by the exhaust ports. The engine piston is of the cross-head type and, by means of a stationary abutment with a gland for the piston rod, a cushioning cylinder is arranged below the piston at C.

Of similar design, the opposed-piston unit is virtually self-balanced, and the engine has the more satisfactory end-to-end scavenging action. The upper side of the lower compressor delivers air to chamber D, whence part is taken for scavenging and charging the engine, and the remainder is passed through the shallow annular passage E to the mixing chamber F, thereby cooling, and absorbing heat from, the combustion zone of the cylinder. Air from the outer ends of both compressors passes through conduit G to the mixing chamber into which the lower side of the upper compressor discharges directly. For a complete installation a plurality of compressor aggregates would be employed. Although the compressors work at only a relatively low pressure, the large number of automatically operated admission and discharge valves would scarcely be regarded with favour by aircraft engineers, particularly as a high engine speed is essential for the treatment of large volumes of air.

An early project of the Junkers Co. employed a multi-bank, radial, two-stroke engine to drive an axial compressor. Referring to Fig. 2 : 16, the air is forced by the compressor A into passages B between opposite pairs of cylinder banks, from which a portion of the supply is taken as scavenging and charging air for the cylinders. The remainder passes round the cylinders as a cooling medium into the mixing chambers C arranged between the cylinder banks alternate to the passages B. Into these chambers the exhaust gases are delivered to mix with the by-passed air, which has been pre-heated by the cylinder walls. From these chambers the mixture expands to the outer air by way of nozzles D. To enable the propulsive effort to be increased, combustion chambers E, into which additional fuel can be introduced through injectors F, are provided between the mixing chambers and the nozzles. The air flow through the complete unit is shown diagrammatically in the plane projection of a longitudinal section of the plant.

In 1935 Hans von Ohain, then a student at Göttingen University, patented a novel turbojet unit having a single-stage centrifugal compressor and a single-stage radial inflow turbine. Backed by the Heinkel aircraft company in 1936, he built a demonstration unit to operate on hydrogen as a fuel. This unit was run on the test bed in March 1937 and produced a thrust of 550 lb. By 1939 a liquid-burning flight unit HeS 3B, weighing slightly less than 800 lb and delivering a static thrust of 1,100 lb had been developed. The specific fuel consumption was about 1·6 lb/hr/lb. It was installed in a specially designed He 178 airframe and Capt. Warsitz made the world's first turbojet flight at Rostock on August 24, 1939. That was a straight flight; circuits were flown on August 27.

The general arrangement of this turbojet unit (Fig. 2 : 18) was



A. Compressor.
B. Air intake passages.
C. Mixing chambers.
D. Discharge nozzles.
E. Supplementary combustion chambers.
F. Supplementary fuel injectors.

Fig. 2 : 16—Junkers jet reaction plant employing a multi-bank, air-cooled, two-stroke engine to drive the rotary compressor. Additional fuel may be injected into the final expansion chambers to increase the propulsive effort.

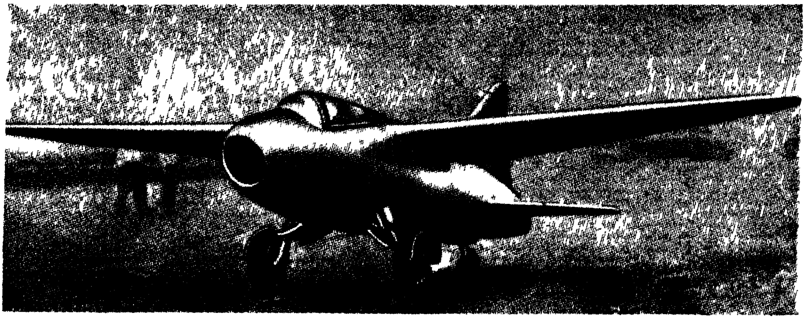
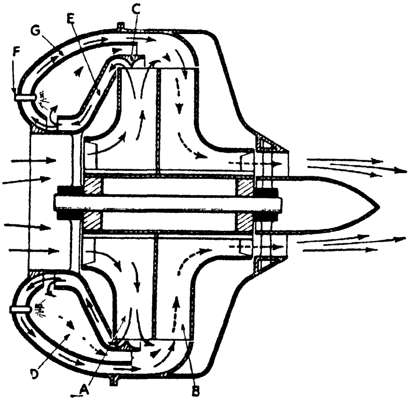


Fig 2 17—The Heinkel He 178, the world's first turbojet aircraft, had a maximum speed of 435 m p.h.



A	*Centrifugal compressor*
B	*Radial inflow turbine*
C	*Air flow guide ring*
D	*Annular combustion chamber*
E	*Combustion air passage*
F	*Fuel sprayer*
G	*Insulation passage*

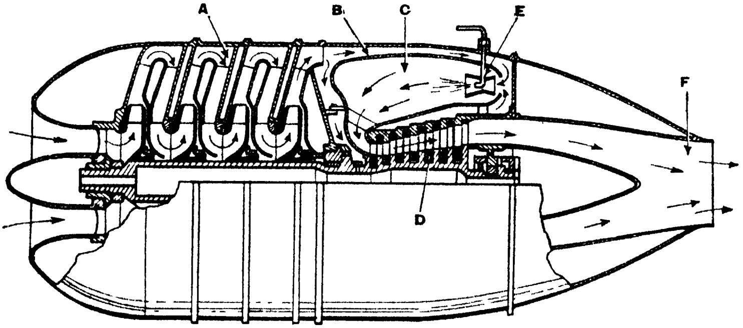
Fig 2 18 A von Ohain unit of the type shown powered the Heinkel 178 which made the world's first turbojet flight in 1939.

revealed in a patent dealing specifically with the combustion chamber, granted to Max Hahn, an associate of von Ohain. A later version, HeS 8A, developed 1,320 lb thrust Two of these were fitted to the He 280 experimental fighter which was first flown in April 1940 and later achieved a speed of about 500 m p.h. In a circular casing having an axial air intake at the front and co-axial discharge orifice at the rear, a shaft is mounted in two bearings supported on spiders. On this shaft is a single rotor which is divided to form a box-type centrifugal air impeller A and a boxed centripetal turbine wheel B. Air drawn into the unit through the forward central aperture leaves the impeller at its periphery and the stream is here divided by a projecting guide C on the enshrouding wall of the annular combustion chamber D.

The main supply is diverted to the rear to the turbine wheel, whilst a smaller portion passes along passage E between the main casing and the combustion chamber wall. At the point of smallest diameter, surrounding the intake aperture, this portion of the air-stream is again divided. Part enters chamber D into which fuel is injected by nozzles F and burned to furnish additional heat for the final jet. The remainder continues around the combustion chamber in space G and joins the initial air-stream and the effluent from the combustion chamber to form a common stream which expands through the turbine and furnishes the motive power for rotating the blower. By passing a portion of the air-stream completely round the combustion chamber this is effectively insulated and heat from the chamber is not completely lost by radiation to the atmosphere but is largely carried off by the air to perform useful work in expansion through the turbine.

Streamlined units for wing installation, to the designs of A. Lysholm, were suggested by Milo A.B., of Stockholm. In one of these, shown in part-section in Fig. 2 : 19, the air enters by an annular orifice, is compressed in a multi-stage centrifugal compressor A and delivered to chamber B. This houses an annular combustion chamber C and the multi-stage turbine D which drives the blower. The air, preheated in its passage over the walls of the combustion chamber and the turbine stator casing, at the rear reverses its direction of flow and enters the combustion chamber past fuel nozzles E. From the combustion chamber the flow is again reversed, and the gases expand through the turbine and pass with an accelerating velocity through an outlet conduit of diminishing cross-sectional area to the discharge nozzle F.

Early editions of this book gave some prominence to the work of S. Campini in Italy. It is of historical interest to retain some account of his early efforts and put on record the performance of the C.C.2, the aircraft

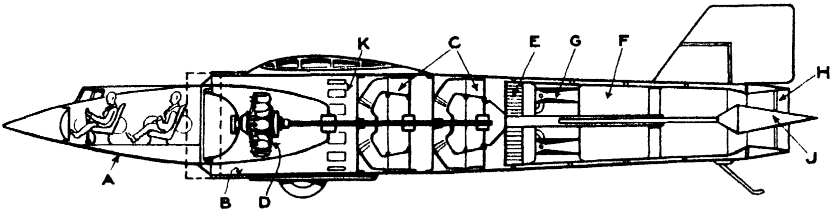


A. Four-stage centrifugal compressor.
B. Air chamber.
C. Combustion chamber.
D. Turbine.
E. Fuel injector.
F. Discharge nozzle.

Fig. 2 : 19—The streamlined Milo unit intended for wing installation. All the air compressed by the 4-stage compressor is passed through the annular combustion chamber.

which first flew in August, 1940, and was widely publicized in 1941 after its Milan-Rome flight.

The C.C.1 design, which never materialized, is shown in Fig. 2 : 20. The project was somewhat optimistically described as being intended for operation at either subsonic or supersonic speeds. The cabin A is of ovoid form and constructed as a pressure unit for high-altitude operation. Air is admitted where the annular space between cabin A and the enshrouding cylinder B has its smallest cross-section. This ram-compressed air then passes through a two-stage centrifugal compressor C driven by a radial engine D or, alternatively, a gas turbine. Beyond the compressor it passes a radiator E, which also acts as a rectifier, into the wide combustion space F. Here, in an annular channel G of venturi section, the air is supplied with fuel and after expansion discharges through the nozzle H. The nozzle unit is arranged to swivel and the cross-sectional area may be varied by means of the ' bullet ' J. On starting, there will be a negative pressure in the inlet chamber between the cabin and the enshrouding ring, and air can be admitted by way of lateral orifices K. The air-admission arrangement outlined above is that employed for operation at subsonic speeds. If the speed of flight is increased to a supersonic velocity a shroud ring is moved forward beyond the former mouth, as indicated by the dotted lines. The air now enters a funnel which first converges and then diverges, thereby taking account of the peculiar conditions of flow which are associated with supersonic velocities.



A. Ovoid cabin.
B. Enshrouding cylinder.
C. Two-stage centrifugal compressor.
D. Radial engine.
E. Rectifier-radiator.
F. Combustion space.
G. Annular mixing channel.
H. Discharge nozzle.
J. Nozzle control bullet.
K. Controlled lateral orifices.

Fig. 2 : 20—The Campini C.C.I design of high-altitude craft intended for operation at either sub- or supersonic speeds. The control cabin is pressure charged. A piston engine is used to drive the compressor.

The C.C.2, which made the celebrated Milan-Rome flight, is now known to have weighed 9,250 lb and to have been fitted with an Isotta-Fraschini liquid cooled engine of 900 h.p. The span was 52 ft. The Isotta engine drove a three-stage ducted fan and ' afterburning ' equipment in the form of a vaporizing burner could be provided in the rear of the fuselage. The moving blades of the ducted fan could be varied in pitch and the area of the exit orifice was adjustable by a movable ' bullet.'

In view of the conjectures concerning the performance of the C.C.2 the following figures may be of interest. The maximum speed was 205 m.p.h. at 9,800 ft, but this could be increased with afterburning to about 230 m.p.h. at the same height. The climb to 13,000 ft took 53 min.

By reason of its high power output, light weight and small bulk the simple jet propulsion turbine offered immediate advantage for military aircraft in the fighter, interceptor and photo-reconnaissance classes which outweighed the handicap of a higher specific rate of fuel consumption than the piston engine. It is only when operated at high aircraft speed at high altitude that the turbojet economically excels the reciprocating engine driving an airscrew. Furthermore the propulsive efficiency of a jet is lower than that of an airscrew at low forward speeds, particularly at take-off. Therefore it appeared unlikely that there would be good prospects for the application of the turbojet to commercial aircraft until the advance of aerodynamic knowledge made higher flying speeds practicable and until cabin pressurization was well established. For operation at medium speeds and altitudes—say 300-400 m.p.h. and 20,000-25 000 ft—it was realized quite early that a turbine driving an airscrew could with advantage bridge the gap between the operating characteristics of the pure jet and the piston engine. In view of recent intensive development of this type, the earlier design projects are of more than passing interest.

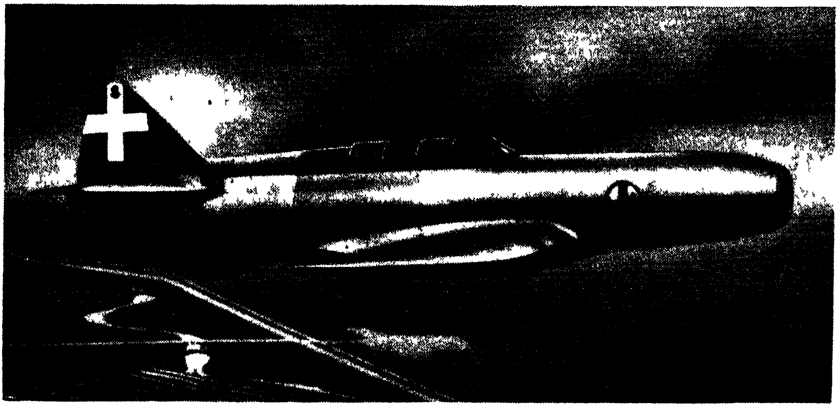
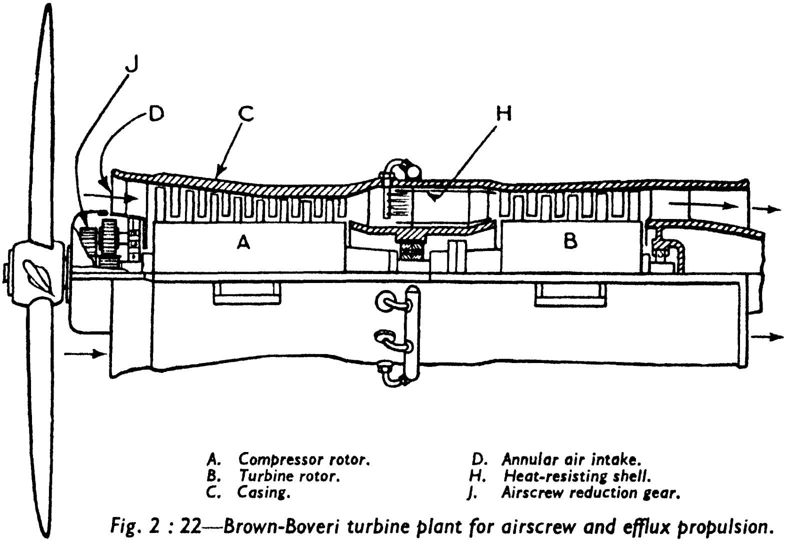


Fig. 2 : 21—The Caproni-Campini jet-propelled experimental craft in the air. Known as the C.C.2, it weighed about 9,250 lb.

Brown-Boveri applied for a German patent in 1939 for the plant shown in Fig. 2 : 22, which was claimed to be of relatively simple construction, low weight and high efficiency. Outstanding features conducing to this end were (1) the arrangement by which the gas flow between compressor and turbine undergoes no substantial deflection or change in velocity ; (2) the diminution of the axial velocity of the air in the compressor to that required for satisfactory combustion ; (3) the provision of small passages serving as mixing chambers in the annular space between the compressor and turbine, and (4) the increase of the axial velocity of the gas as it flows through the turbine.

Compressor rotor A and turbine rotor B are on directly connected co-axial shafts, and the stator blades of both components are mounted in a tubular casing C forming the main duct. Air enters at the annular inlet D and is compressed in the usual manner in the forward (convergent) section of the compressor. Thereafter the duct diameter is parallel and the angle of the rotor blades is decreased. Only 20 to 30 per cent of the air delivered by the compressor is required for the combustion of fuel ; the remainder is employed to lower the temperature of the combustion gases.



A. Compressor rotor.
B. Turbine rotor.
C. Casing.
D. Annular air intake.
H. Heat-resisting shell.
J. Airscrew reduction gear.

Fig. 2 : 22—Brown-Boveri turbine plant for airscrew and efflux propulsion.

The smaller quantity of air is passed into a plurality of separate mixing chambers E in the annular space, as shown in the plane development diagram Fig. 2 : 23. In these the air is further reduced in velocity and mixed with the fuel injected through nozzles F. Flame forms at or beyond the grid bars G, which furnish a number of friction surfaces along which the boundary layers of the mixture move at lower velocity than the ignition velocity. Small flames are formed locally which preheat and ignite the main mass of the mixture and thus maintain stable conditions. The flaming mixture is reduced in temperature by the excess air which flows through the passages between the mixing chambers. To

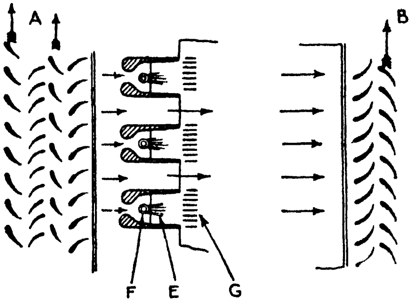
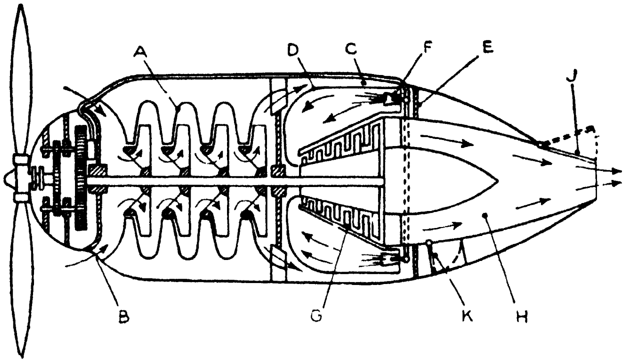


Fig. 2 : 23—Developed section of Brown-Boveri annular combustion space with mixing chambers and flame grids.

A. Compressor blading.
B. Turbine blading.
E. Mixing chambers.
F. Fuel nozzles.
G. Flame grids.

protect the walls of the annular space a shell H of heat-resisting sheet metal may be fitted. Cooling air passes between the shell and the walls. Although this plant is primarily intended to drive an airscrew through reduction gear J, the turbine effluent is employed to give a supplementary propulsive effort by jet reaction. The turbine, therefore, differs from the usual type, which is of increasing (divergent) diameter to reduce the gas velocity and exploit the gas expansion in order to minimize exhaust losses. To be effective for propulsion, it was suggested, the gas outlet velocity should be about 1·2 to 1·5 times the speed of the aircraft. Accordingly, the turbine is substantially of parallel diameter and the rotor blade angle is progressively increased towards the outlet to raise the gas velocity at the discharge nozzle.

Another turboprop unit, patented in 1936 by the Swedish firm Aktiebolaget Ljungströms Angturbin, is shown diagrammatically in Fig.



A. Centrifugal compressor.
B. Air inlet.
C. Annular air duct.
D. Combustion chamber.
E. Bulkhead.
F. Fuel nozzles.
G. Gas turbine.
H. Discharge duct.
J. Discharge flap.
K. Effluent by-pass flap.

Fig. 2 : 24—Ljungströms Angturbin turboprop unit with multi-stage centrifugal compressor.

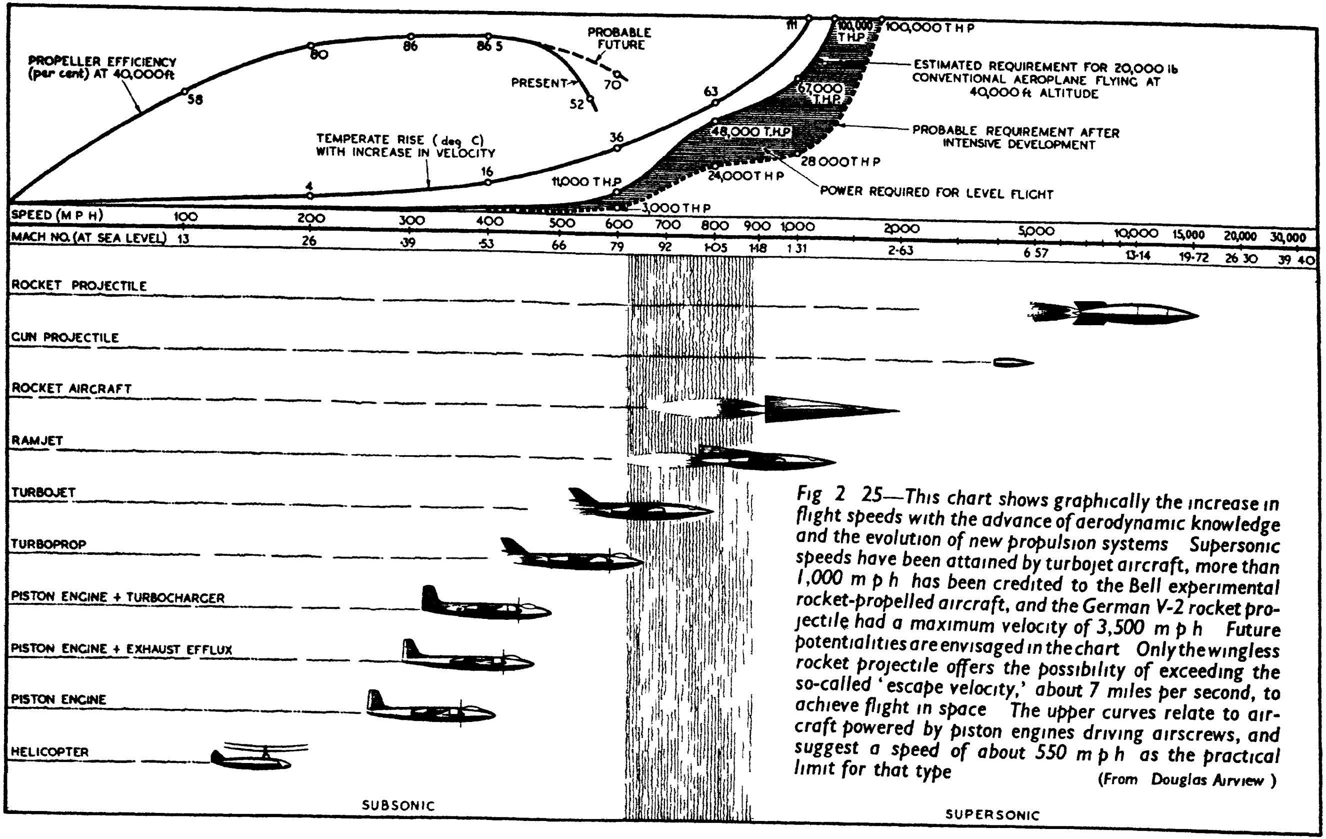


Fig 2 25—*This chart shows graphically the increase in flight speeds with the advance of aerodynamic knowledge and the evolution of new propulsion systems Supersonic speeds have been attained by turbojet aircraft, more than 1,000 m p h has been credited to the Bell experimental rocket-propelled aircraft, and the German V-2 rocket projectile had a maximum velocity of 3,500 m p h Future potentialities are envisaged in the chart Only the wingless rocket projectile offers the possibility of exceeding the so-called 'escape velocity,' about 7 miles per second, to achieve flight in space The upper curves relate to aircraft powered by piston engines driving airscrews, and suggest a speed of about 550 m p h as the practical limit for that type*

(From Douglas Airview)

2 : 24. The air is compressed in a multi-stage centrifugal blower, and to reduce overall length the annular combustion chamber envelopes the turbine. Particular importance is attached to increasing rapidly the propulsive effort for take-off and in emergency and special control arrangements are provided for this purpose. Housed in a streamlined casing, the four-stage centrifugal compressor A draws in air through inlet B and delivers to a space C surrounding the annular combustion chamber D. At a bulkhead E the air stream is reversed and enters the combustion chamber past a plurality of shrouded fuel nozzles F. Combustion occurs in the vicinity of the nozzle shrouds and the temperature of the combustion gases is lowered by the excess air. At the forward end of the combustion chamber the flow is again reversed to enter the gas turbine G. From the annular outlet of the turbine the relatively low-velocity effluent flows along a convergent duct H which raises the velocity at the nozzle.

Normal regulation of aircraft speed is effected by adjustment of supply of fuel at nozzles F. For rapid acceleration, however, a sudden and substantial increase of fuel would raise the gas temperature unduly as the compressor cannot be instantaneously accelerated to deliver a corresponding supply of air. To facilitate this speed-up, the power input to the compressor is temporarily increased by reducing the power applied for propulsion of the aircraft either by the airscrew or the efflux reaction or both. Subsequently the proportionate division of energy between compressor and propulsion is restored to normal at the higher level, and it is claimed that this operation is considerably quicker than the method of gradually increasing the fuel supply to the desired amount. For this purpose a flap J in the discharge duct may be raised to increase the area of the discharge orifice. Alternatively or additionally, a flap K adjacent to the turbine outlet may be lifted to by-pass a portion of the effluent. By either or both means the propulsive effort is lowered, the back pressure on the turbine is reduced giving an increased head and pressure drop in the turbine, which thus produces more mechanical energy which is entirely absorbed by the compressor. Another method of achieving the desired results, either instead of or in addition to decreasing back pressure, is to increase the slip of the airscrew by temporarily fining the pitch.

CHAPTER 3

Thrust and Performance

THRUST is an applied force tending to produce motion in a body or to alter the motion of a body. Horse-power is a measure of energy expended in a given period of time. The mechanical energy delivered by a piston engine or a gas turbine can be measured in terms of horse-power on the test bed by a water brake, an electric dynamometer or a torquemeter. This, however, is merely an expression of the energy available and is no criterion of the individual or comparative effect in propelling the aircraft. It is necessary to know the method by which the power will be applied and the efficiencies of the respective propulsion systems employed.

Whether using an airscrew or a jet the aircraft obtains its forward propulsion by reaction to a mass of air being accelerated rearwards. Relatively, in the case of the 12 ft- to 16 ft-diameter airscrew a larger mass of air is projected rearwards at a lower speed, whilst with the jet a smaller mass, say a column about 1 ft diameter, is accelerated to a higher velocity (see Fig. 3 : 1). Thrust is the ' equal and opposite ' reaction to the force producing this change of momentum and exerts a forward push on the aircraft. The term ' push ' is used deliberately for even a tractor airscrew pushes against a thrust collar in the engine nose in a similar manner to a horse which pushes against the collar of the harness when it is commonly described as pulling a cart.

The thrust horse-power, t.h.p., of a piston engine-airscrew propulsion unit is the shaft horse-power, s.h.p., of the engine multiplied by the efficiency of the airscrew, say 80 per cent at sea level and 400 m.p.h. On a turboprop unit energy is divided between the airscrew and the jet, so the available s.h.p. of the turbine after driving the compressor must be multiplied by the aircrew efficiency and an addition must be made for the supplementary thrust of the residual jet. The whole of the power developed by the turbine of a pure jet propulsion unit is absorbed in driving the compressor and all thrust is obtained from the jet.

It is only possible to compare the propulsive effect directly in terms of thrust. The efficiencies of different types under varying conditions of operation, however, do not follow similar curves. As a consequence, a direct comparison can only be made in terms of thrust under identical operating conditions. This involves such variables as aircraft speed, altitude, atmospheric conditions, temperature, airscrew efficiency, jet efficiency and aircraft drag. Certain of these factors are of such importance that they must not be regarded merely as modifying or auxiliary influences but as first-order effects.

It would, for example, be unrealistic to ignore the effects of forward speed and altitude on the functions of a jet unit. Similarly, though for different reasons and to different effect, it would be unrealistic to disregard them in connection with a conventional piston-engine airscrew unit for aircraft propulsion. This, however, does not preclude or invalidate investigation of the piston engine under static, sea level, test-bed, laboratory or even hypothetical conditions.

A brief examination of some basic features of a jet propulsion unit, whilst unrepresentative of actual performance, will be of interest. First, it is desirable to ascertain what energy is available in a specific plant and resort is made to the standard formula to determine a force :

$$F = M \times A$$

in which F is the force, M the air mass and A the acceleration. Mass cannot be determined directly but must be ascertained by reference to weight, W, which is the measure of the force of gravity. Now, gravity is the force which causes a freely falling mass to accelerate towards the centre of the earth at the precisely determined rate of $32{\cdot}18$ ft/sec^2, denoted by ' g.' By substitution, therefore, the formula can be restated as :

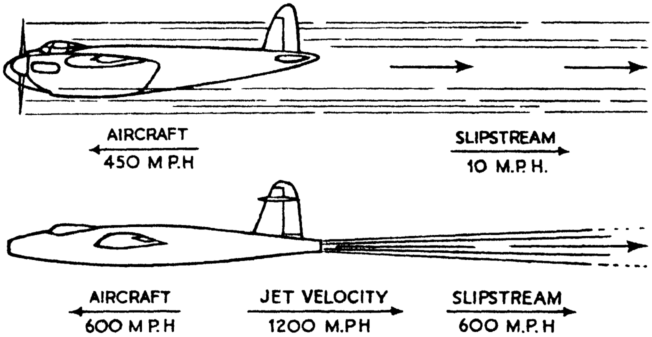


Fig. 3 : I—Both airscrew-driven and jet aircraft are propelled by reaction of a mass of air projected rearwards.

$$W = M \times g$$

and again as :

$$M = \frac{W}{g}$$

THRUST

Presuming quiescent atmospheric conditions, the air mass enters the unit at an intake velocity, IV, equal to the forward speed of the aircraft and is accelerated to the jet velocity, JV, at which it is discharged. The change of velocity, CV, effected in the power unit is the difference between IV and JV. When the aircraft is stationary IV will, of course, be zero. To obtain thrust, T, the standard formula can now be expressed :

$$T = M \times CV$$

or alternatively :

$$T = \frac{W \times CV}{g}$$

By way of example, in a unit having an air mass flow of 60 lb/sec and a jet velocity, relative to the aircraft, of 1,760 ft/sec (1,200 m.p.h.) the static thrust will be :

$$T = \frac{60 \times 1{,}760}{32{\cdot}18} = 3{,}281 \text{ lb}$$

Assuming JV is constant, the thrust of the unit will vary according to the variation of CV or in other words according to the forward speed of the aircraft. Obviously, thrust is highest when the aircraft is stationary and there is maximum acceleration of the air mass, whilst if it were possible for the aircraft speed, and consequently the intake velocity IV, to equal the jet velocity JV, there would be no thrust whatever. This would suggest that thrust falls directly with increase of aircraft speed, as indicated in the dotted curve of Fig. 3 : 2. In practice, however, thrust will fall until the aircraft speed reaches 300–350 m.p.h. Thereafter, increased speed operates to advantage due to ram pressure rise in the intake duct of the compressor. Thrust will rise to regain its static maximum at about 650–700 m.p.h., as shown in the full curve of Fig. 3 : 2, and will continue to rise until the aircraft reaches at least sonic velocity. Beyond that the rate of increase of thrust may be expected to lessen as a result of shock losses in the intake duct and compressor entry. The vital importance of ram pressure will thus be appreciated.

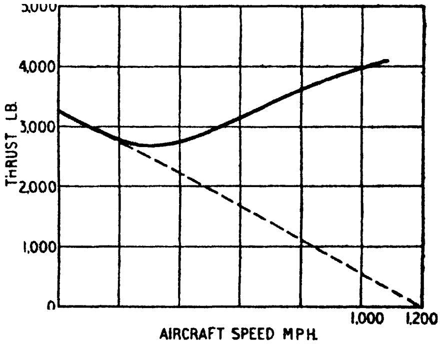


Fig. 3 : 2—Thrust tends to fall with increased aircraft speed but soon rises under influence of ram pressure.

THRUST HORSE-POWER

Thrust T, at any particular aircraft speed, may be converted to thrust horse-power, t.h.p., by multiplying the thrust by the speed of the aircraft and dividing by the appropriate factor to obtain standard horse-power units of 33,000 ft-lb/min or 550 ft-lb/sec. If thrust is expressed in lb and aircraft speed in ft/sec the product is ft-lb/sec and the formula is :

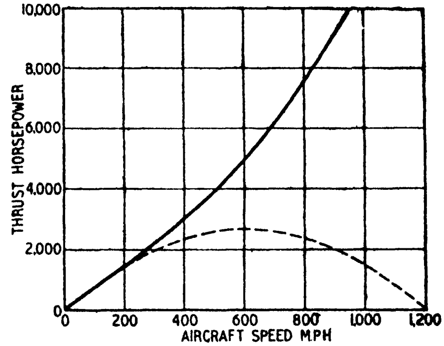


Fig. 3 : 3—Thrust horse-power rises as aircraft speed is increased.

$$\text{t.h.p.} = \frac{\text{T} \times \text{IV}}{550}$$

An alternative would be to obtain mile-lb/hr by multiplying thrust in lb by the aircraft speed in m.p.h. when the formula would be expressed :

$$\text{t.h.p.} = \frac{\text{T} \times \text{IV}}{375}$$

When the aircraft is stationary no energy is expended on propulsion and should the aircraft travel at the same speed as the jet velocity JV again no propulsive work is done. If thrust horse-power is calculated from the dotted thrust curve (Fig. 3 : 2) it will be found to rise from zero when the aircraft is stationary, to a maximum when the aircraft speed is half the jet velocity, and then fall at a similar rate as the aircraft speed is increased to that of the jet velocity. This is plotted in Fig. 3 : 3 in dotted line together with a more realistic curve calculated from the full line thrust curve of Fig. 3 : 2.

PROPULSIVE EFFICIENCY

Assuming again a constant jet velocity, it will be found that the propulsive efficiency of the jet directly depends upon the ratio, R, of the jet velocity to the forward speed of the aircraft. Obviously, it is zero when the aircraft is stationary and 100 per cent when the aircraft speed is the same as the jet velocity. Knowing the jet velocity, the

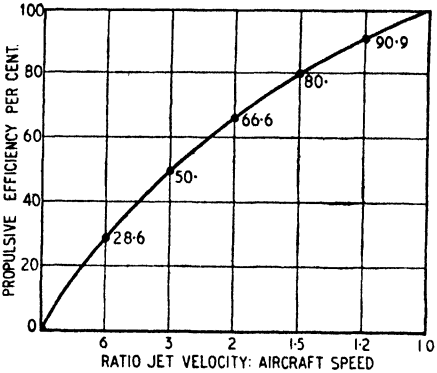


Fig. 3 : 4—Propulsive efficiency rises as the ratio of jet velocity to aircraft speed is reduced.

efficiency at any aircraft speed can be determined by:

$$\eta = \frac{2}{R + 1}$$

and the complete range is plotted in Fig. 3 : 4. This shows a remarkable contrast to the characteristic curve of airscrew efficiency which falls from about 80 per cent at 400 m.p.h. to 70 per cent at 500 m.p.h. and 50 per cent at 600 m.p.h. Thus, at speeds above 500 m.p.h. the jet becomes the more efficient means of propulsion. High speed is doubly advantageous to the jet propulsion unit as it improves ram pressure and increases the propulsive efficiency.

Optimum performance is obtained with a relatively short jet pipe. When an extension pipe is necessary to meet requirements of installation, performance is adversely affected. On the de Havilland Ghost, for instance, thrust is reduced 0·24 per cent and specific fuel consumption increased 0·32 per cent approximately for every foot length of the extension pipe. Jet velocity can be raised and thrust increased by reducing the diameter of the jet nozzle. In one unit, the de Havilland Goblin, an additional 250 lb thrust was obtained by reducing the nozzle diameter from 16¼ in to 16 in. There is an adverse effect, however, as reduction of diameter increases back pressure on the turbine. This leads to a sharp rise in operating temperature and consequent danger of damage to the turbine blades and disc.

EFFECT OF ALTITUDE

With aircraft operating at high instead of low altitude, drag is reduced and less propulsive effort is needed. It is estimated that the drag of a particular aircraft at 40,000 ft is only approximately one-third the value at sea level. Altitude, however, has a number of conflicting influences on the component efficiencies of aircraft propulsion units. The reduction of air temperature with increasing altitude improves compressor efficiency, or in other words less mechanical energy is required to obtain a specific compressor output. Of course, the higher the speed the greater the temperature rise as the air in the duct leading to the compressor is accelerated to aircraft speed. At 500 m.p.h. this may produce a rise of 25° C, but if the aircraft is flying at, say, 30,000 feet the atmospheric fall will be about 60° C yielding a net drop of 35° C.

Output will fall due to the reduced density of the air and this tends to reduce the power delivered by the turbine. Conversely, owing to lowered temperature and pressure behind the turbine, there is a better expansion ratio through the turbine which increases its efficiency. If the rate of fuel supply was maintained, gas temperature at the turbine nozzle would rise, the turbine would accelerate and the jet velocity would be increased. In practice a device sensitive to barometric pressure reduces the fuel delivery as altitude is increased to maintain a constant turbine speed of rotation at the desired operational rating—cruise, climb or combat.

Of course, turbine output can be increased by raising the gas temperature at the turbine nozzle and, within the capacity of the turbine metals, this expedient can be employed on a turboprop unit. Unless aircraft speed can be increased to maintain the jet velocity : aircraft speed ratio, however, it is liable to be disadvantageous on a turbojet unit as a higher jet velocity means a lower propulsive efficiency as too much of the energy would be going into the slip stream.

It has been shown that whilst the efficiencies of the piston engine and the airscrew decrease with higher speeds and altitudes the turbojet enhances its efficiency. There remains an aerodynamic factor adversely affecting airscrew propulsion. At high altitudes and consequent low temperatures, sonic velocity is reduced (Fig. 3 : 5) and thus problems of compressibility will arise at lower aircraft speeds. From about 760 m.p.h. at sea level, sonic velocity falls to about 660 m.p.h. at 40,000 ft.

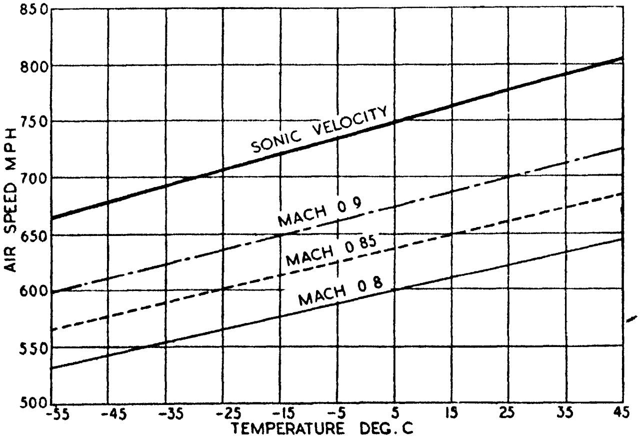


Fig. 3 : 5—The speed of sound varies with air temperature. Mach value is proportional to sonic speed and varies accordingly. The onset of compressibility is delayed by high atmospheric temperature but advanced at high altitude due to the temperature lapse rate.

At speeds approaching the critical Mach number of the aircraft there will be a steep increase in drag with either airscrew or jet systems. On an airscrew, however, a further complication arises. As the linear air speed of the airscrew blade tips is a combination of rotational and forward speeds, compressibility burble at the blade tip, which will drastically reduce the airscrew efficiency, is likely to occur.

OPERATING TEMPERATURES

A gas turbine resembles a reaction-type, non-condensing steam turbine, in that a fluid medium under heat and pressure is expanded through stationary and rotating blades to produce mechanical energy. In a gas turbine there are smaller pressure and temperature differences but the velocity of the medium is many times greater, which indicates why the rotor blades of a gas turbine are of aerofoil section instead of the crescent moon section of steam turbine blades.

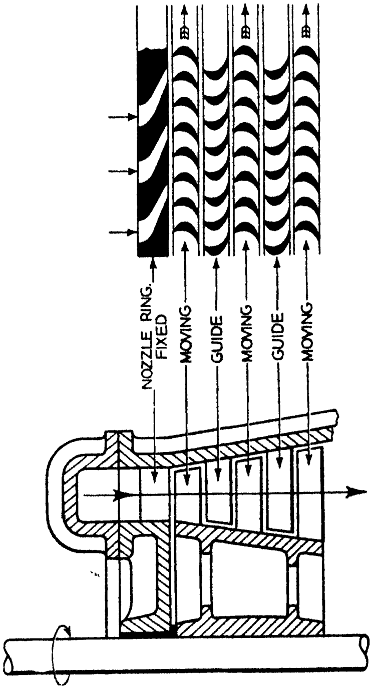


Fig. 3 : 6—Diagram of steam turbine with solid nozzle ring and crescent-shaped rotor and stator blades.

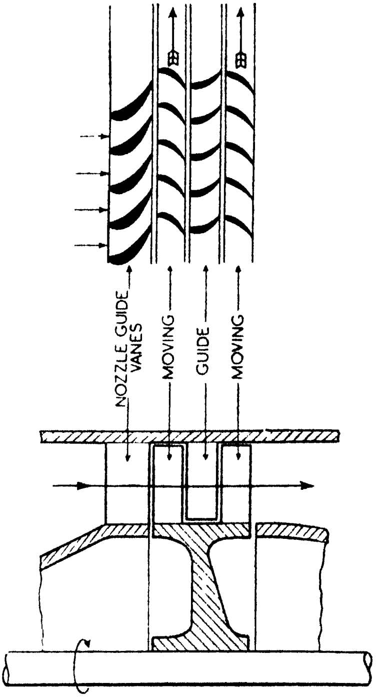


Fig. 3 : 7—Rotor and stator blades of a gas turbine are of aerofoil shape. Nozzle passages are defined by guide vanes.

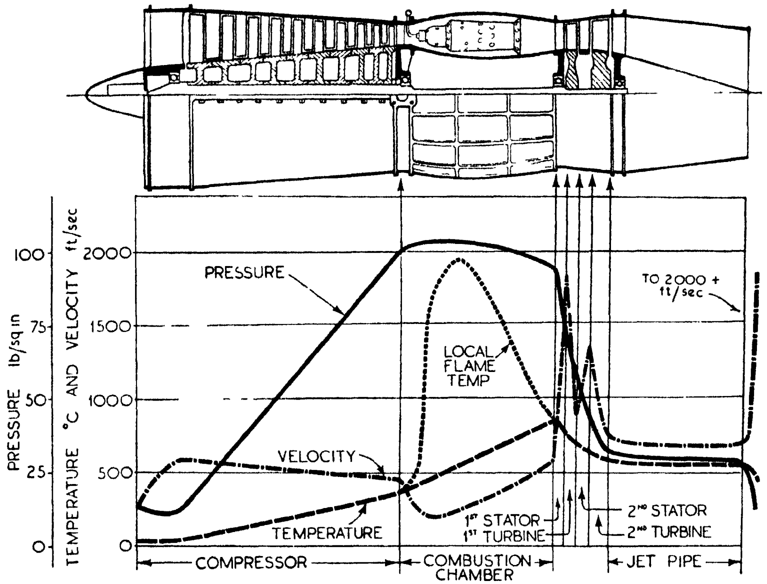


Fig. 3 : 8—Curves showing variations of pressure, temperature and velocity of the gas flow through a simple turbojet.

Several times more air passes through the system than is necessary for complete combustion of the fuel, the excess air being utilized to dilute the combustion products and keep the temperature at the turbine entry down to working limits In a modern turbine, gas pressures are of the order of 105 lb/in^2 abs (90 lb/in^2 gauge). High temperatures are imperative for efficiency but a limit is imposed by the capacity of the materials of the turbine to withstand the heat in operation. It must be appreciated that high performance without long working life is unacceptable save in special types of military aircraft.

Apart from the actual combustion zone the highest temperature occurs at the turbine inlet, say 850° C, so that in the case of a multi-stage turbine only the initial row of blading operates under maximum temperature conditions. Since the gas is cooled in the process of expansion at each stage, it may emerge at half the original temperature in a four-stage unit. With a single turbine wheel, the temperature of the jet stream at the nozzle exit would be much higher, say 650° C.

At the cost of some loss of simplicity the efficiency of a gas turbine unit can be improved by the use of intercoolers between compressor stages, heat exchangers to preheat combustion air or reheaters between turbine stages. Considerable improvement in performance can be obtained by ‘afterburning.’ In this system additional fuel, from atomizers located to the rear of the turbine, is burned to increase the expansion rate through the final jet nozzle. An increase of thrust exceeding 50 per cent has been attained. As the fuel is burned under conditions of low pressure and the effect is to increase jet velocity, and therefore to lower propulsive efficiency, it is relatively wasteful. Its application will consequently be confined to short-period operation at take-off and in emergency. A further opportunity of thrust ' augmentation ' is provided by the ducted fan or tunnelled airscrew. Taking energy respectively from the gas stream leaving the turbine or from the turbine shaft such systems supply an additional mass flow of air at a lower velocity to supplement the main jet. In Fig. 3 : 9 is shown the ducted fan of the Metropolitan-Vickers F-3 unit. This subject is also dealt with in Chapter 4.

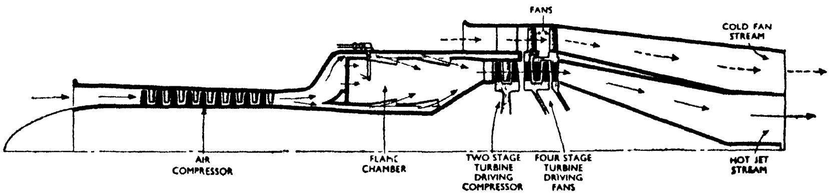

FANS
COLD FAN STREAM
AIR COMPRESSOR
FLAME CHAMBER
TWO STAGE TURBINE DRIVING COMPRESSOR
FOUR STAGE TURBINE DRIVING FANS
HOT JET STREAM

Fig. 3 : 9—This diagrammatic section shows the flow path of the Metrovick F.3 turbojet unit with ducted fan augmenter.

COMPONENT EFFICIENCIES

In 1939 a paper was read by Dr. Adolf Meyer on ' The Combustion Gas Turbine ' before the Institution of Mechanical Engineers. During the discussion which followed, F. Whittle (at that time a Squadron Leader of the R.A.F.) made a notable contribution to the subject. He remarked :

' So far little has been said about the gas turbine in relation to aircraft, which is regarded as a most hopeful field for it, for as the gas turbine was taken up into the air the reduced atmospheric temperature made possible very much higher efficiencies than could be obtained on the ground. Using a simple formula for the thermodynamic efficiency of a gas turbine, and the usual symbols :

$$\text{Overall efficiency} = \frac{\eta_t K - \dfrac{1}{\eta_c}}{K - 1} \times \text{air standard efficiency}$$

where K was the ratio of positive to negative work in the ideal cycle and was therefore equivalent to the ratio of maximum temperature to compression temperature in the ideal cycle ; η_t was the turbine efficiency, and η_c the compressor efficiency. This formula was slightly defective because it made no allowance for the fuel saved due to the heating effect of the losses in the compressor, neither did it allow for (*a*) variable specific heat, or (*b*) the fact that the mass flow through the turbine was greater than the mass flow through the compressor by the amount of the liquid fuel injected into the combustion chamber. Otherwise, however, it gave a close approximation to the overall efficiency of the gas turbine.

‘ The formula could be re-written :

$$\eta_0 = \frac{\eta_t\eta_c K - 1}{(\eta_c K - 1)} \times \text{air standard efficiency}$$

in which the relative importance of turbine and compressor efficiencies was more clearly shown and it could be seen that for a given value of the product $\eta_t\eta_c$ the overall efficiency was inversely proportional to the compressor efficiency.

‘As an example, assuming a turbine efficiency of 80 per cent and compressor efficiency of 80 per cent adiabatic, $\eta_t\eta_c = 0{\cdot}64$. With a compression ratio of 6·35 and a maximum combustion temperature of 1,100° C abs, the overall efficiency worked out at 19·67 per cent ; whereas with a turbine efficiency of 90 per cent and compressor efficiency of 70 per cent the overall efficiency was 22·8 per cent for the same compression ratio and maximum temperature. Thus, though $\eta_t\eta_c$ was smaller, the overall efficiency was greater. In the calculations, allowance was made for the heating effect of the compressor losses. Of course the machine with the lower compressor efficiency would require either more stages in the compressor, or higher blade speed to obtain the same pressure ratio. He therefore agreed with the implication that the turbine efficiency was relatively much more important than the compressor efficiency.

‘ The temperature of the international standard atmosphere decreased by 2° C for every 1,000 feet of height and therefore for a given maximum temperature and a given compressor speed (i.e. a given adiabatic temperature rise of compression) both the air standard efficiency and the ratio *K* of positive to negative work would increase owing to the lower initial air temperature.

‘ For example, with a turbine efficiency of 85 per cent and compressor efficiency of 70 per cent, a maximum temperature of 1,100° C abs, and an adiabatic temperature rise in the compressor of 200° C, the comparative figures between sea level and 35,000 feet were :

	K	*Thermo-dynamic efficiency, per cent*	*Air standard efficiency, per cent*	*Overall efficiency, per cent*	*B.h.p. per lb of air, per sec*
Ground level ..	2·258	45	41	18·45	59·3
35,000 feet	2·630	56·5	47·85	27	98·5

Thus there was an increase of 46 per cent in the overall efficiency, and of 66 per cent in the power per pound of air per second, on rising from the ground to the stratosphere. Of course the total power fell off, due to the much reduced air density, but so also did the power required for a given flying speed. The overall efficiency of an internal combustion engine, on the other hand, fell off to such an extent that an unsupercharged engine would scarcely be able to turn itself round against its own mechanical losses at heights of the order of 40,000 feet. At 35,000 feet, i.e., the beginning of the stratosphere, the air temperature was about 218° C abs.

'The big flying boat had reached a point where it was limited in development by its power plant; at present no suitable engine would produce more than 2,000 h.p. in one unit, although power units with an output of 5,000–10,000 h.p. were needed. Moreover, there were, as was well known, strong arguments in favour of flying at very great heights. The combination of these two factors resulted in a demand in the aeronautical world for efficient gas turbines.

'Some speakers appeared to be pessimistic about turbine efficiencies, and the figure of 85 per cent had been discussed. He had never been able to find a satisfactory reason why efficiencies should not be higher than that. If the methods of the aeronautical engineer were applied to turbine design, he could see no fundamental reason why turbine efficiency should be below 94 per cent, taking account of the boundary layer theory and other factors.'

An interesting survey, made by the Swiss firm Sulzer Bros., of the theoretical possibilities of a variety of component combinations in an open cycle turbine plant is presented graphically in Fig. 3 : 10. It will be appreciated that thermal efficiencies realizable in practice depart very materially from those quoted, and consequently the figures have primarily a comparative value. None of the components is 100 per cent efficient, and thus some proportion of the total energy released is lost in the operation of each unit. For the complete plant the practical efficiency may be only from 40 to 60 per cent of the theoretical efficiency, depending on the cycle selected. All examples are worked out for a cycle pressure ratio of 4 : 1 and cycle temperatures of 20° C minimum and 648° C maximum.

(1) The simple basic cycle with adiabatic compression, a single combustion chamber and a single turbine with adiabatic expansion.

(2) The simple cycle modified by one stage of intercooling to give an approach to isothermal compression. Expansion remains adiabatic.

(3) By introducing a second stage of combustion an approach is made towards isothermal expansion.

(4) Reverts to simple adiabatic compression and expansion but a recuperative heat exchanger is added to transfer the maximum theoretical

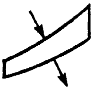
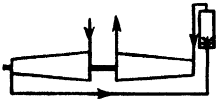
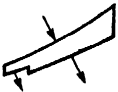
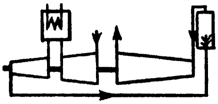
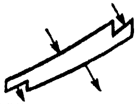
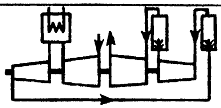
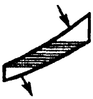
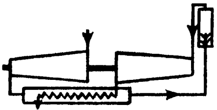
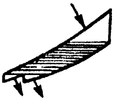
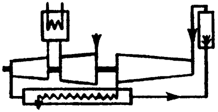
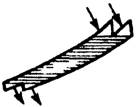
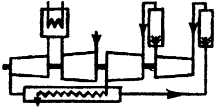
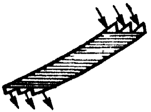
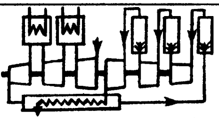
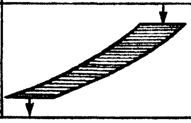
ENTROPY DIAGRAM		DIAGRAMMATIC ARRANGEMENT	THEORETICAL THERMAL EFFICIENCY (PER CENT)
1			31·5
2			30·0
3			34·5
4			53·2
5			58·2
6			61·6
7			63·0
8		IDEAL CYCLE	68·2

Fig. 3 : 10—Various arrangements of the open cycle gas turbine plant and the theoretical thermal efficiency of each. All are based on a pressure ratio of 4 : 1 and a gas temperature range of 20° C to 648° C.

quantity of heat from the exhaust gases to the compressed air before it enters the combustion chamber.

(5) Intercooling is beneficial in a cycle that includes recuperation.

(6) Further improvement in efficiency is obtained by an additional stage of combustion.

(7) Two stages of intercooling and three stages of combustion make

compression and expansion more nearly isothermal and give the closest approach to the ideal.

(8) Indicates the ideal cycle in which compression and expansion are purely isothermal and there is total recuperation of heat from the exhaust gases. This would yield the maximum theoretical efficiency of 68·2 per cent for the postulated cycle pressures and temperatures.

An increase of the pressure ratio would considerably influence the theoretical efficiencies of cycles without recuperation but would exercise progressively less influence on more complex cycles. The relation between practical and theoretical efficiency is lower with cycles having adiabatic compression and expansion and is more marked in such cycles as the pressure ratio is increased. As compression and expansion approach nearer the isothermal the proportional loss between the practical and theoretical efficiency does not increase so rapidly with increase in pressure ratio. More complex cycles, as (6) and (7), can attain the maximum practical efficiency at a considerably higher pressure ratio than 4 : 1, but an increase in pressure ratio will not markedly raise the theoretical efficiency.

CHAPTER 4

Jets or Airscrews

At their inception turbojet aircraft acquired exceptional prominence by reason of their novelty and the high speeds attained. Understandably, an impression gained ground that the gas turbine was essentially confined to propulsion by jet reaction. That system is clearly of major importance but the cardinal achievement was the development to the stage of practical operation of a prime mover more powerful, less bulky and of lower specific weight than the highly developed and universally adopted piston engine of the period. Gas turbine power units may utilize jet reaction or may drive an airscrew or ducted fan for the propulsion of aircraft.

A reciprocating engine can never attain the smoothness of operation of a perfectly balanced rotating mechanism such as the turbine, and cannot be operated at rotational speeds even approaching those practicable in a turbine. No matter which working cycle is employed and however many cylinders are incorporated, a piston engine has intermittent power impulses and must be more prone to vibration and torsional oscillation than a constant-pressure gas turbine.

For a turbojet power plant the basic components are a compressor, delivering air to a combustion chamber in which liquid fuel is burned, and a turbine producing shaft power solely to drive the compressor. Of the potential energy of the fuel consumed approximately two-thirds is utilized by the turbine and one-third remains for the propulsive jet. In a simple turboprop unit the same three basic elements are employed but the heated gas is expanded in the turbine to a wider ratio, that is, to a lower

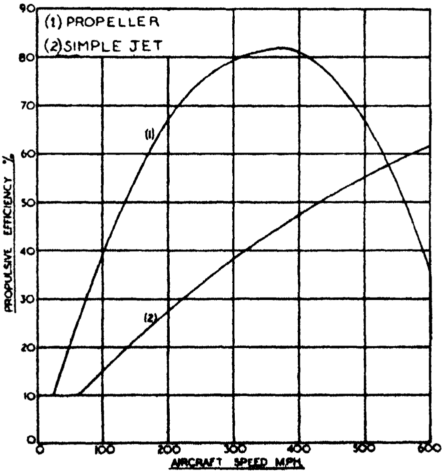


Fig. 4 : 1—The propulsive efficiency of a jet is inferior to that of an airscrew at the lower aircraft speed ranges.

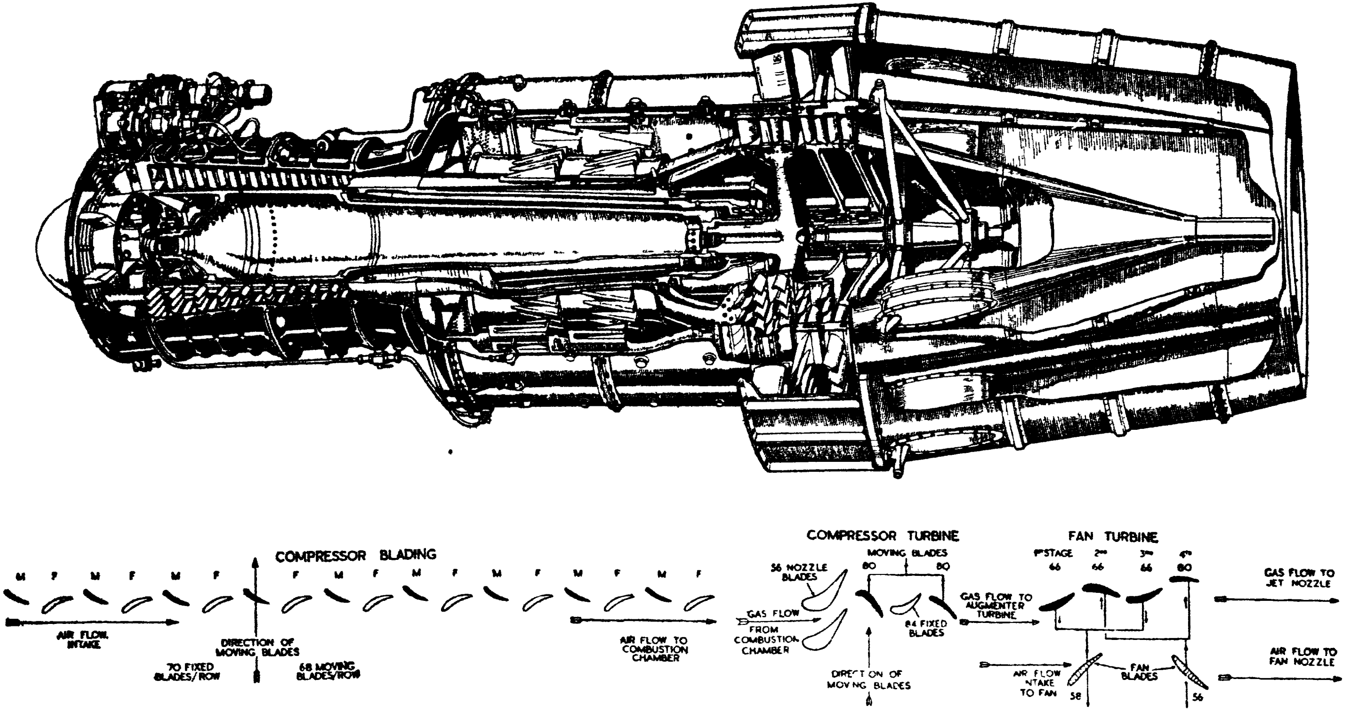
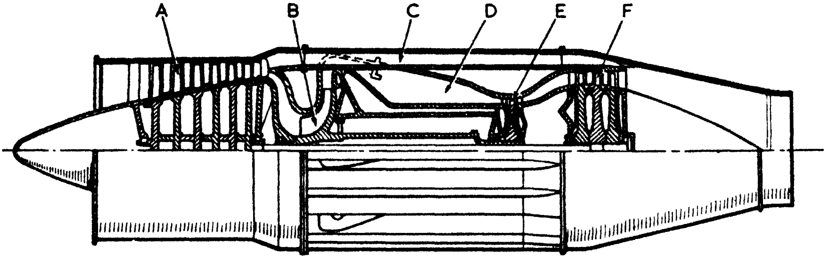


Fig. 4 : 2—Sectioned arrangement and blading diagram of Metrovick F 3 ducted-fan unit.

pressure. Shaft power is thus produced in excess of that required for the compressor and is available to drive a propeller. Energy remaining in the turbine exhaust can be employed as a supplementary propulsive jet.

The division of energy is not in a fixed proportion and the unit is designed to give a thrust ratio to meet the operational needs of a specific aircraft or type of aircraft. Since propulsive efficiency rises as the ratio of the velocity of the air mass displaced rearwards to the forward velocity of the aircraft is reduced, see Fig. 3 : 4, it follows that the lower the aircraft operational speed, the higher the proportion of energy that will be taken from the gas stream for propulsion by airscrew.

Between the turbojet and turboprop types may be considered the ducted-fan and by-pass units. While giving propulsive efficiencies superior to the pure jet for take-off and climb they avoid the limitations of the airscrew at higher speeds. Thus, their operational field may be estimated to be in the range of aircraft speeds from, say, 450 to 550 m.p.h. It is, however, unsatisfactory to group power units into limited speed classes as the characteristics of the various types assume varying importance in relation to the operating requirements of the aircraft in which they are installed.



A. Axial compressor.
B. Centrifugal compressor.
C. By-pass duct.
D. Combustion chamber.
E. H-p turbine.
F. L-p turbine.

Fig. 4 : 3—De Havilland scheme for by-pass unit with independently driven axial and centrifugal compressors.

In the fan-augmented unit some energy in a turbojet is taken to drive a fan that takes in air, raises its pressure slightly, and passes it through a duct to the rear. It is discharged, at an enhanced velocity, either to join the gas stream behind the turbine or directly to atmosphere to envelop the main jet. The fan may be at the front and driven from the compressor or at the rear and driven by an independent turbine, as in the Metrovick F.3 (Figs. 3 : 9 and 4 : 2).

There is a similar division of the air stream in the by-pass unit. All the air taken in is given an initial low compression and then part is ducted to the rear, by-passing the turbine, and part is further compressed and delivered to the combustion chamber or chambers in the usual manner. The by-pass air may be taken from an intermediate stage of an axial compressor or from the delivery of a l-p unit while the air for the turbine is further compressed in an independent h-p compressor. In Fig. 4 : 3 is shown a de Havilland scheme for a by-pass unit with an axial l-p compressor driven by a three-stage l-p turbine and a centrifugal h-p compressor driven by a single-stage h-p turbine.

In both fan-augmented and by-pass types thrust can be boosted temporarily by burning additional fuel in the by-passed air as in the manner of the turbojet afterburner.

Selection of the propulsive system is largely determined by the conditions under which the unit is to operate. The governing factors are speed, operating height, payload and range. The ultimate criterion is not the specific efficiency of the power plant but the overall efficiency of the complete aircraft.

The influence of the gas turbine on aircraft design merits separate treatment, but it may be noted here that due to its relatively small dimensions, smooth contour and light weight, a turbine unit may be conveniently submerged in the wing or fuselage of an aircraft with extremely beneficial effect on aerodynamic design. This improvement will apply in different degree whether jet propulsion or airscrew propulsion is employed, but the advantage is more marked in the case of pure jet propulsion. This factor is of such importance that, despite the lower thermal efficiency of the gas turbine compared with an orthodox reciprocating engine, it is possible for a jet-propelled aircraft to have a greatly superior performance. Owing to the relatively high rate of fuel consumption, however, it will not be superior in range to a piston-engined craft. Only the compounded unit, with either a reciprocating engine or

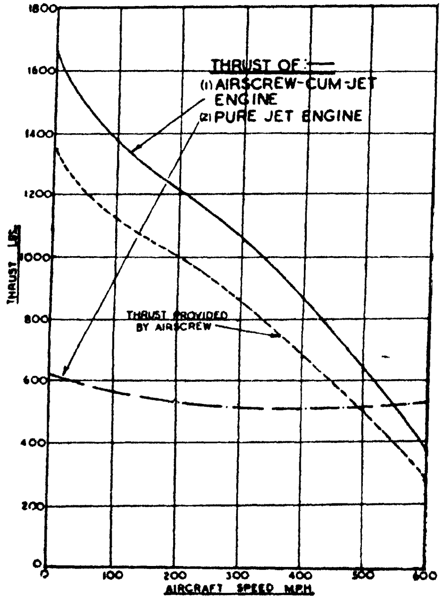
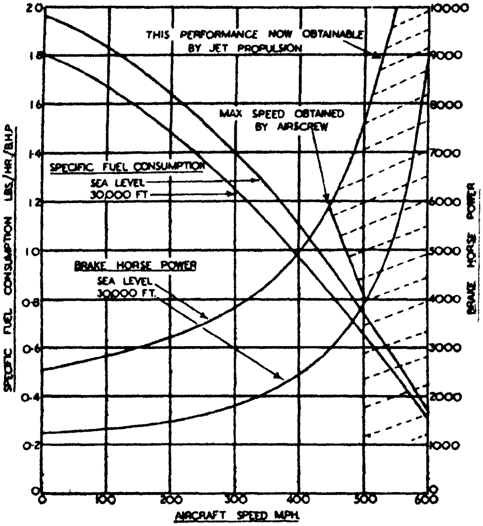


Fig. 4 : 4—Comparison of pure jet and airscrew-jet units. The gas horse-power of both units under static conditions is 1,000.

Fig. 4 : 5—Performance of jet unit (4,500 lb static thrust) expressed in equivalent piston engine form. (Dr. Hooker.)



free-piston gasifier supplying a power turbine, can challenge the piston engine in that respect.

At low aircraft speed at low altitude a jet unit has relatively poor thermal and propulsive efficiencies. It follows that as regards take-off and initial rate of climb a jet-propelled aircraft is inferior in performance to a comparable aircraft employing airscrew propulsion. To overcome this handicap at take-off, thrust augmentation systems have been proposed for jet units. These take the form of water or methanol injection at the compressor (wet compression) or the injection of fuel in the gas stream behind the turbine (re-heating or afterburning). B.M.W. in Germany fitted a liquid rocket unit on the jet pipe, with propellant pumps driven from the jet unit, to obtain 100 per cent thrust increase.

As aircraft speed increases, the propulsive efficiency of the jet efflux improves. As higher altitudes are reached air temperature falls, the compression ratio increases, and the thermal efficiency improves. On a thrust horse-power basis the specific rate of fuel consumption falls with increasing forward speed, whereas on a direct thrust basis the specific rate rises with increasing forward speed Since high forward speeds can be attained only by expenditure of enormous energy, the gross or actual fuel consumption in unit time will increase with forward speed.

Since the rate of fuel consumption of a turbojet is particularly high at full throttle and low altitudes, there must be a severe limitation on the range of an aircraft unless load is reduced. These facts indicate that for some time to come pure jet propulsion will be selected for high-speed, medium altitude, short-range aircraft or special high-speed transports operating at high altitude.

Here it is appropriate to quote from a 1943 observation in an earlier edition of this work : ‘If large passenger transports with pressurized cabins flying at 500 m.p.h. at very great altitudes are the aim, then jet *propulsion will almost certainly be employed. On the other hand, if opinion is confirmed that transports of the future will cruise at about* 250 m.p.h. at the lower altitudes, then turbine-driven airscrews may well be chosen for propulsion. It must, of course, be presumed that the rate of fuel consumption can be improved as development work proceeds, and that metallurgists can assist by the production of better heat-resisting materials for the turbine blades.'

Technical development in the interim has raised the optimum speeds from 500 to 550 m.p.h. and from 250 to 450 m.p.h. respectively but, apart from that, the foregoing statement is still substantially correct. The de Havilland Comet turbojet liner and the Vickers Viscount turboprop passenger transport have been introduced on scheduled airline services in several countries.

Two factors led to the prior development of the turbojet. They were :

(*a*) Military demand for the highest possible aircraft speeds ;

(*b*) Concentration by manufacturers on the simplest form of power unit in order that the problems associated with turbines might be understood and overcome.

So far turboprops have been aerodynamically and thermodynamically somewhat inferior to the best turbojets. The position is now being redressed and it may be expected that turboprops with efficiencies approaching or equalling those of turbojets will become available. These will have a still lower rate of fuel consumption and will undoubtedly widen the range of turboprop operation.

The addition of a reduction gear and an airscrew to a turbine adds considerably to its weight and thus to some extent cancels out the great advantage of the turbine power unit, which is its low specific weight. However, the conception of the constant-power turboprop, that is, one having a full-throttle height of 20,000 to 30,000 ft, makes it possible to design a unit with, proportionately, a much lighter gear and a very much lighter airscrew.

For example, assuming an output of 3,000 h.p. at 30,000 ft was required, a conventional unit would develop approximately 10,000 h.p. at sea level and a reduction gear and propeller to absorb that power would be very heavy and very large. On the other hand, four engines of 10,000 h.p. would be sufficient to take off an aircraft weighing nearly 400,000 lb, whereas transport or civil aircraft of the foreseeable future are unlikely to weigh more than 200,000 lb and will only require four units each developing 4,000 to 5,000 h.p. at take-off. It follows that the the ideal turboprop is one that maintains constant power from sea level to as high an altitude as possible, and for the immediate future that constant power should be of the order of 4,000 to 5,000 h.p.

The adaptability of the airscrew for use as a brake is a point of great importance. Airscrew braking reduces the wear on tyres and removes the danger of a tyre bursting under emergency-stop loads. It is an essential on icy runways where wheel brakes are ineffective. The airscrews can also be used to increase the gliding angle on approach to an aerodrome and thus reduce the danger of under- or over-shoot. Finally, the airscrew has the great merit of giving a very high thrust for take-off and of having a very high efficiency under cruising conditions.

It has been stated that the overall noise level in an aircraft with turbojet units is 30 decibels below that in a machine with equivalent piston engines and airscrews. This, however, must be regarded as an empirical statement. The noise of a piston engine is highest at low frequencies and will decrease as the frequency is raised to about 10,000 cycles/sec. Conversely, jet engine noise is low at low frequency and increases to a high level at about 30,000 cycles/sec. Such a frequency is beyond the aural point of human beings but that does not imply that these effects can be entirely ignored. In the air the normal sound-proofing of a pressurized cabin furnishes adequate insulation and protection from such frequencies. The turboprop has proved to be considerably quieter in operation than the turbojet.

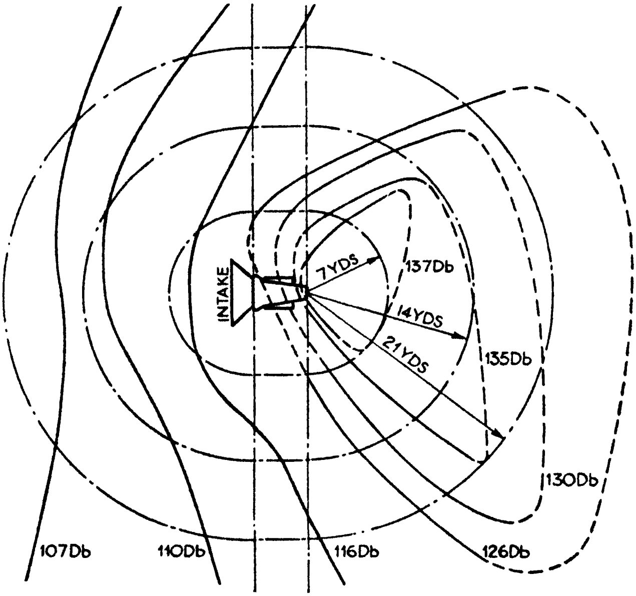


Fig. 4 : 6—Distribution patterns of noise around static Nene jet unit running at 11,000 r.p.m.

As with piston engines, noise is more objectionable to persons in the vicinity of the aircraft than to the passengers when engines are being run up prior to take-off. In Fig. 4 : 6, prepared by Dr. E. W. Still, B.Sc., are plotted the distribution curves of noise around a Rolls-Royce Nene static jet unit running at 11,000 r.p.m. He suggests that a radius of 100 ft from the engine is necessary to ensure a comfortable level of noise at take-off.

Counter-rotating or co-axial airscrews, tractor or pusher type, with as many as five blades to each unit, should to some extent lessen the designer's problem of providing adequate blade area with a reduced diameter and operation at higher rotational speeds without exceeding the permissible helical speed of the blade-tips. They possess the further advantages of allowing a shorter and lighter undercarriage, and of smoothing out the slipstream and consequently improving the air flow over the control surfaces. If pusher-type airscrews are employed, interference with air flow by the effects of the slipstream can be obviated.

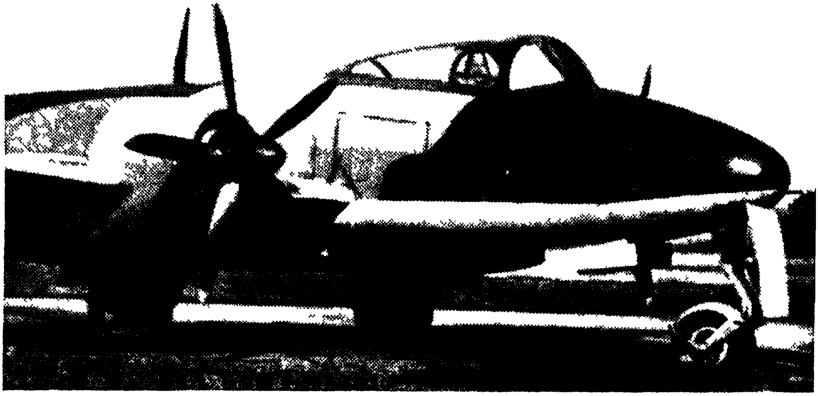


Fig. 4 : 7—The first turboprop aircraft ever to take the air was the Gloster Meteor on 20th September, 1945. The two Rolls-Royce Trent units had five-bladed airscrews.

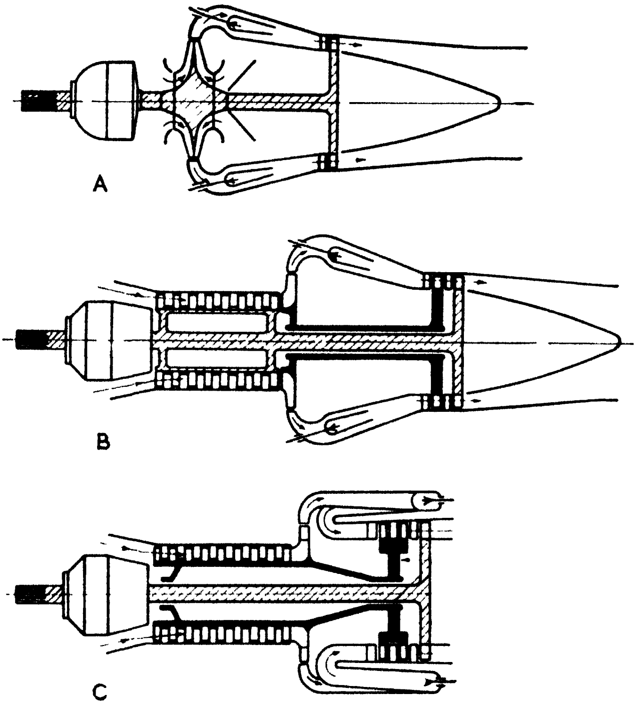
Airscrews for gas turbines will duplicate in essential design those for equivalent piston engines, but the absence of vibration and torsional oscillation will permit the weight to be decreased by using thinner blades and smaller root sections than hitherto.

All the foregoing references relate to airscrews as at present in production. Turbine propulsion has brought higher aircraft speeds, higher rotational speeds and higher operating altitudes and together these factors have tended to advance the onset of compressibility and offset the gains of improved design. Consequently, it is still a reasonable assumption that the airscrew can maintain a practical efficiency up to, say, 450 m.p.h., and will necessarily be superseded by jet propulsion for higher aircraft speeds.

Studies are being made, however, of a variety of blade shapes and sections in attempts to produce an airscrew that can be operated at supersonic helical velocities. Some progress has been made experimentally with blades of swept-back shape and also with square-tipped, rectangular blades of ultra-thin, knife-edged section. Should these researches lead to practical developments the situation will be changed and the scope of airscrew propulsion could be extended to sonic aircraft speeds.

A DESIGNER'S VIEWS

Airscrew design is a highly specialized subject and, as in previous editions, Mr. L. G. Fairhurst, M.I.Mech.E., F.R.Ae.S., chief engineer



A. Direct-connected.—Turbine drives compressor and airscrew.

B. Compounded-compressor.—First turbine stage drives free centrifugal compressor. Second turbine stage drives axial compressor and airscrew.

C. Free-turbine.—First and second turbine stages drive free compressor. Third turbine stage drives airscrew.

Fig. 4 : 8—Basic types of airscrew-turbine.

of Rotol Ltd., has contributed the following authoritative review of current practice and trends :

Airscrews for reciprocating engines have for some time been fully established with such features as constant-speeding, feathering and, more recently, reverse pitch as a landing brake. The use of the propeller in a ‘wind-milling’ pitch for dive-braking on the dive-bomber has been experimented with on the reciprocating engine, but to date has not found any definite application on this type of engine. Gas turbine power units have created additional problems for the airscrew designer which were not experienced hitherto on reciprocating engines, and have demanded a completely new range of airscrews for outputs of from 1,000 h.p. to 7,000 h.p. The fact that there are at present three distinct basic types of turboprop units has contributed by no mean amount to existing airscrew design problems. These consist of the ‘direct-connected,’ ‘compounded-compressor’ and ‘free-turbine’ types shown in Fig. 4 : 8. The characteristics of each type demand special mechanical features in the airscrew design. Such features and associated aerodynamic, vibration, and noise-reduction problems are dealt with subsequently under the various engine types. One feature common to all power units having a central air intake is the reduction in bulk of the airscrew hub in order to house it inside the smaller diameter spinner associated with this type of intake. This requirement in itself involved considerable re-design by having to cramp the operating mechanism into a smaller compass than hitherto.

DIRECT-CONNECTED TURBINE

This type of power unit demands, in addition to the normal fine pitch angle for take-off, a pitch below this angle for starting purposes. The requirement can better be appreciated by reference to Fig. 4 : 9,

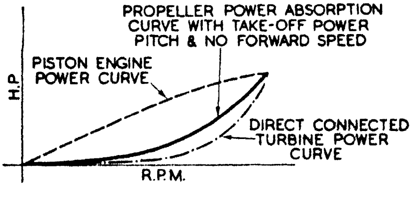


Fig. 4 : 9—Comparative power curves of piston engine and direct-connected turbine under static conditions with airscrew in take-off power pitch.

which shows that the piston engine under static conditions is capable of starting and accelerating the airscrew with the latter set at take-off pitch angle. The direct-connected turbine curve, falling below the airscrew power absorption curve, shows that it cannot start and accelerate the airscrew at normal fine-pitch angle as its power only equals the airscrew power absorption at maximum r.p.m. Thus, it is necessary to provide an additional pitch angle, below normal fine, which is usually between 0° and +6°. Furthermore, it is essential to interconnect the airscrew pitch and engine fuel control as otherwise, if too low an r.p.m. be selected on the airscrew for a given engine power after starting, the airscrew will overload the engine and result in stalling and overheating. Fig. 4 : 10 shows the relation between the engine and airscrew on a sudden acceleration of the engine from slow running to maximum r.p.m. As the airscrew control would be set to call for maximum r.p.m., no change from starting pitch angle would occur until the engine slightly overspeeds beyond its maximum r.p.m. at which point the pitch angle would then have to coarsen rapidly in order to hold the permissible maximum r.p.m. figure. Under such conditions some over- and under-swing about maximum r.p.m. is unavoidable. The latest development consists of an additional feature in the existing form of constant speed unit in the nature of an acceleration-conscious control, which enables the airscrew pitch to be progessively changed by a signal from acceleration or deceleration of the engine thus reducing the degree of over- and underswing of r.p.m. on sudden throttle openings. Under normal flight conditions the basic constant speed unit functions in the usual manner.

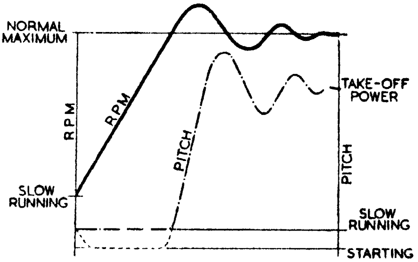


Fig. 4 : 10—Relation of airscrew speed and pitch under rapid acceleration from slow running to maximum r.p.m.

The need for close association between airscrew pitch and fuel control makes it essential to extend the constant speed range down to starting pitch to guarantee a correct and automatic relation between engine and airscrew. If the airscrew is of the reverse-pitch type with provision for a negative angle of around 25°, it becomes necessary further to extend the constant-speed range down to this figure and also to reverse the sense of the constant-speed control coincident with passing from positive to negative pitch.

Having now accommodated in the airscrew normal fine, engine-starting and reverse pitches it becomes essential to provide automatically removable safety locks at the normal fine and starting pitches in order to prevent the airscrew blades ' fining off ' to too low a pitch in the event of airscrew control or engine failure in flight. Either failure combined with an airscrew moving suddenly into starting pitch would not only seriously over-speed the propeller and engine but would also impose on the aircraft a very heavy drag from the airscrew. On a non-reversing airscrew the sequential operation of the automatic safety lock is as follows : The engine is initially started with the propellers in super-fine pitch (from 0° to 6°), the blades remaining in this pitch until starting has been accomplished. On opening up the engine to full power and r.p.m. the blades coarsen until they reach the take-off fine-pitch angle associated with such power and r.p.m. At the same time they ' latch over ' the safety lock which remains made throughout flight and until the aircraft has again landed. At the instant the undercarriage legs are loaded, the lock is automatically removed by electro-hydraulic means and the airscrew can now constant-speed between normal fine and starting pitch according to the taxying conditions required.

While engine designers generally are striving to make their engine stalling characteristics such that the normal fine-pitch lock need not be removed until after touch-down it may, for the time being, be essential on certain types to break the lock on the approach.

In the case of a reversing propeller, a second lock is fitted at starting pitch which can also be automatically removed after the aircraft has landed and the reverse pitch control operated.

In order quickly to reduce airscrew drag at the moment of engine failure at critical take-off speed, airscrews are now having an automatic-coarsening pitch feature incorporated. The auto-coarsen feature takes effect at around zero engine torque, being initiated from the engine torque meter. The feature is only operative during take-off, that is, with the throttle levers in the full forward position. A signal alternative to that from the torque meter can be obtained by the use of pitot heads mounted at the wing leading edge, one within the airscrew slipstream and the other, at the extremity of the wing, being affected only by airspeed pressure. With both engines operating at full take-off power, the slipstream pressure at each airscrew is maintained at more than 2·5 in of water (around 10 in maximum) above the airspeed pressure. Should one engine fail, the differential pressure between airspeed and slipstream falling below 2·5 in initiates the automatic coarsening.

COMPOUNDED COMPRESSOR

In so far as airscrew design is concerned, this type functions on the lines of the piston engine. It is, therefore, possible to start and accelerate the unit up to maximum power and r.p.m. with the propeller set at normal take-off fine pitch. Only when blade pitch angles below normal fine are required, as for braking, is there any necessity to include the automatically controlled safety lock. Aircraft drag induced by a windmilling airscrew after engine failure is less than for an equivalent size of direct-connected turbine, since there is less inertia in the 'half compressor' allied to the propeller.

FREE TURBINE

As in the previous case, this unit does not demand a special fine pitch for starting. Aircraft drag induced by a windmilling propeller after engine failure is only a fraction of that for the direct-connected type and still considerably less than the compounded compressor type. This is due to the very low inertia of the airscrew shaft system, as there is only a turbine wheel to rotate. On account of this low inertia, however, there is a greater tendency to overspeed the propeller and free turbine after failure of the propeller control system, particularly at speeds approaching cruising. To prevent this overspeeding an automatically controlled safety lock can be incorporated just below the cruising pitch against which the airscrew will 'sit' after failure; this pitch being sufficiently coarse to prevent overspeeding. The problem of reduction of noise level arising from airscrew tip speed is somewhat eased in the free-turbine by its relatively easier capability of being temporarily up-speeded for take-off without affecting the cruising r.p.m. already selected for low noise level characteristics. This temporary up-speeding gives considerably worth-while increase in take-off thrust, the order of which is shown in Fig. 4 : 11.

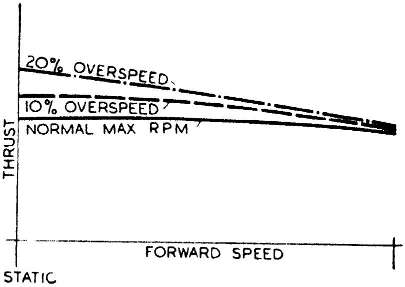


Fig. 4 : 11—Increase in take-off thrust with temporary overspeeding.

AIRSCREWS FOR DIVE-BOMBING AIRCRAFT

The gas turbine will no doubt be the power unit for which the airscrew designer will develop the windmilling pitch feature to give added braking effect in the dive to that provided by the normal diving flaps. The blade pitch angle selected is that at which the propeller is very inefficient as a windmill developing high drag for a small windmilling input power to the engine. On the heavier types of naval dive-bombers the normal diving flap controls have been found to be too heavy both in control load and avoirdupois. It has been shown that on an aircraft of 26,000 lb all-up weight the propeller can provide a braking effort of 14,000 lb in the dive. The three different types of turbine units react differently on the airscrew design for dive braking, since the rate of pitch change into and out of the diving pitch vary with the engine type. The direct-connected turbine with the maximum inertia in the airscrew shaft system requires around 40–50° per second to prevent serious overspeeding when passing through the pitch angle at which the airscrew is a comparatively efficient windmill. As the inertia in the airscrew system is reduced higher rates of pitch change are necessary up to figures of 70–80° per second for the compounded compressor engine and 100–120° per second for the free turbine.

MECHANICAL FEATURES OF TURBINE AIRSCREWS

A typical propeller hub and a circuit design for a turbine unit airscrew are shown in Figs. 4 : 12 and 4 : 13 respectively. The hub drawing reveals the effort made to reduce the height of the hub sockets to enable them to be housed within a small-diameter spinner. The mechanism of the safety locks is shown inside the propeller operating cylinder, being automatically made and removed by employing increased oil pressure from the airscrew controller unit.

The circuit diagram indicates the complete airscrew control system including the reversing feature. It should be noted that electric solenoid control has been resorted to for operating the various change-over hydraulic valves used in the reversing and lock removal mechanism.

FEATURES OF THE GAS TURBINE AFFECTING AIRSCREW AERODYNAMIC DESIGN

A characteristic of the gas turbine is the close ratio between maximum power r.p.m. and cruising power r.p.m. Generally the cruising r.p.m. is 90 per cent of maximum as against 80–85 per cent for the piston engine. For civil aircraft the rotational tip speed of the airscrew is usually reduced to minimize propeller noise. With the single-speed reduction gear, this lowering of the airscrew cruising r.p.m. also lowers the airscrew take-off r.p.m. and hence jeopardises take-off thrust due to greater stalling of the blades. This effect is more pronounced on the gas turbine power unit than on its piston engine counterpart, due to the closeness of cruising and take-off r.p.m. on the former engine, as referred to at the beginning of this paragraph.

To compensate it is necessary to obtain a greater blade area or activity factor as far as is practicable without affecting cruising performance. The difficulty of compromising between take-off and cruise conditions has been a problem in the industry for many years, and two-speed gears have been mooted at various times. As mentioned earlier, the free turbine has great possibilities in this respect, as temporary overspeeding of the airscrew turbine can be accomplished without affecting the compressor portion of the engine. It provides in effect an infinitely variable gear.

The higher forward speeds projected with gas turbine units have accelerated the development of blade aerofoils of higher aerodynamic

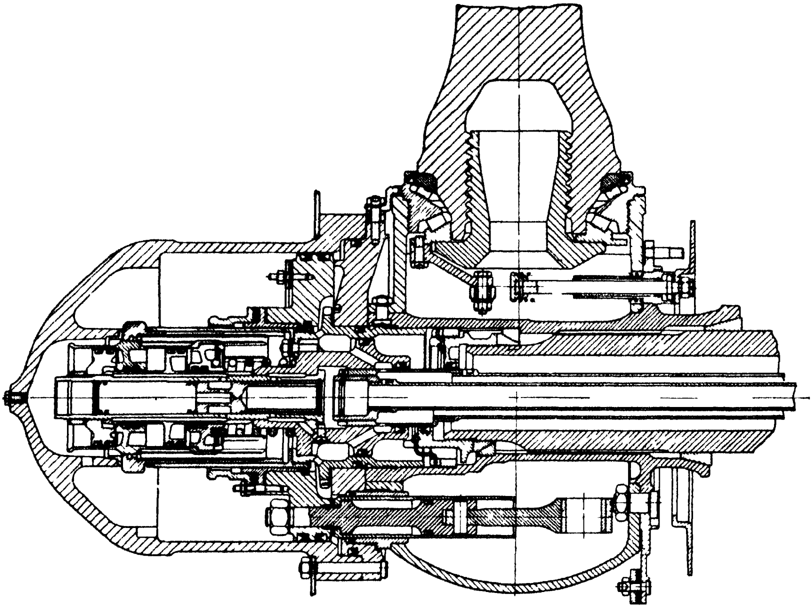


Fig. 4 : 12 Section of typical hydraulically operated airscrew hub. Alternative shafts are shown in upper and lower section halves.

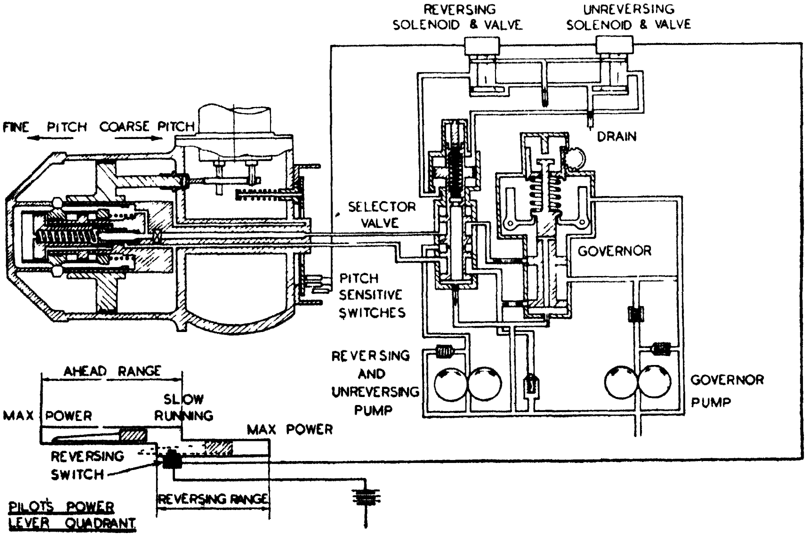


Fig. 4 : 13—Circuit diagram for an airscrew with reverse pitch range.

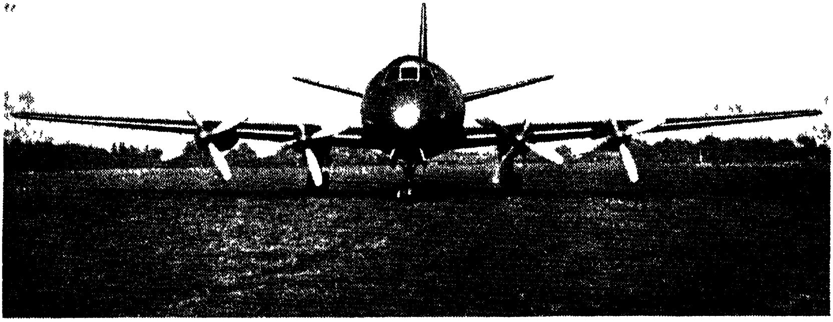


Fig. 4 . 14—The first medium-range turboprop airliner · Vickers Viscount powered by four Rolls-Royce Dart units

efficiency. N.A C.A. Series 16 sections are now being widely used on all the latest turbine airscrew projects and give considerably increased thrusts for take-off and climb and increased cruising efficiency.

Further developments include the swept-back blade. The critical speed at which high drag losses occur in a blade of this type is around 25 per cent higher than hitherto. The use of boundary layer suction on the airscrew blade is receiving very serious consideration as an effective reduction in drag of the section can be thus achieved. A blade of hollow construction lends itself excellently for utilizing the inherent centrifugal 'pumping' effect for sucking away the boundary layer.

AIRSCREW NOISE PROBLEMS

The gas turbine, being in itself a less noisy power unit than the piston engine, has forced the airscrew designer to a serious study of noise level reduction.

Airscrew noise is in two forms, rotational noise and vortex noise. The noise from rotational effects contains the greater acoustic energy and is heard as a low-pitched roar. So far the only method of reduction is to operate at lower cruising r.p.m.. While such r.p.m. reduction gives a most effective lowering of the noise level, the use of a single reduction gear between engine and airscrew seriously jeopardizes the take-off thrust and demands in most cases a compromise. The frequency of the rotational noise is a product of the r.p.m. and the number of airscrew blades.

The vortex noise is caused by turbulent air creating vortices which are shed off the airscrew blades in the vicinity of the tips. So far very little has been or can be done to reduce this form of noise, but since it has by far the less acoustic energy the problem has not the immediate importance of the rotational effect.

Acceptable noise levels laid down for modern aircraft lie between 98 and 100 decibels.

COUNTER-ROTATING AND CO-AXIAL PROPELLERS

The higher powers attainable with gas turbines have brought to the fore the necessity for dual rotation propellers. With power units of 5,000 to 7,000 h.p. counter-rotating or co-axial types are inescapable as the aircraft designer is still reluctant to accommodate diameters of 17 and 18 ft at which single rotation could be retained. It is possible, however, to avoid the complication of counter-rotating and co-axial airscrews by the use of four- or five-bladed airscrews with chordal widths of around 18 in to 20 in. To achieve a reasonable airscrew weight, such blades would essentially have to be of the hollow steel construction. Beyond the 5,000 h.p. mark the present practice is to couple two engines of equal power. The type of turbine determines the choice between counter-rotating or co-axial designs. It is the usual practice to use the counter-rotating arrangement on coupled free-turbines and the co-axial arrangement on coupled direct-connected turbines. This difference in the airscrews occurs because in the case of coupled direct-connected turbines each half of the propeller has to be independently driven by its own engine. Thus, in the event of one engine failing, the good engine does not have to expend 70 per cent of its normal power in driving the dead engine

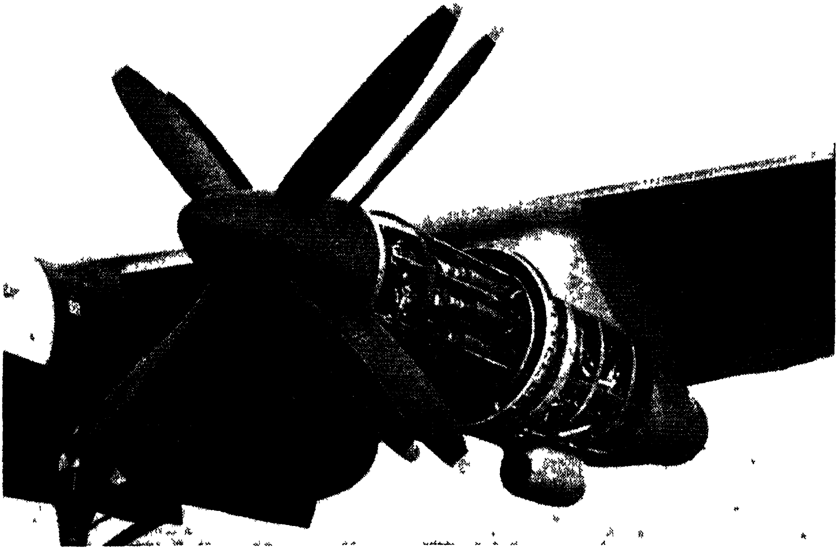


Fig. 4 : 15—Counter-rotating airscrews on Armstrong Siddeley Python.

through the gearing. With the free-turbine, the power to drive the dead engine is relatively negligible.

WEIGHTS OF TURBINE AIRSCREWS

Being lighter in weight than its piston engine counterpart, the gas turbine has inspired the airscrew designer to provide lighter propellers. The greatest saving in airscrew weight is achieved in blades where, on account of the absence of torsional vibrations from the power unit, considerable reduction in the thickness of section is possible. A certain degree of weight reduction can also be achieved in the hub structure, again owing to the absence of torsional stresses.

Whilst the torsional stresses from engine excitations are no longer a problem with the gas turbine engine, the increased speed and wing loading of present-day aircraft has accentuated a first order airscrew vibration which hitherto was of negligible consequence. This 1XP airscrew vibration is induced by the variations in lift forces occurring on each of the blades during each complete revolution of the airscrew.

The primary cause of this lift variation is the angularity of the air flowing through the airscrew disc, which angle is in turn affected by such items as : location of the thrust line of the airscrew with respect to the chord of the wing ; location of the airscrew in front of the leading edge of the wing ; velocity of the aircraft ; and wing loading.

The immediate effect of this lift variation of the blades is to produce a periodic flapping or bending of the airscrew blades with a time frequency for one cycle of the vibration equal to that for one revolution of the airscrew. Since this type of vibration is excited entirely by aerodynamic forces, it is not affected by the reduced vibration characteristics of non-reciprocating engines, such as turbines. As a result, in the design of turbine airscrews, full advantage cannot be taken of the light weight and thin blades theoretically possible with this type of power plant.

Whilst a theoretical method of determining these 1XP stresses has been devised, it is now mandatory to confirm these stresses by flight strain-gauging before an airscrew can be granted its Certificate of Airworthiness.

Specific weights of airscrews for gas turbines vary between 0·15 to 0·18 lb/h.p. for single-rotation and 0·18 to 0·22 lb/h.p. for dual-rotating types.

AIRSCREW PERFORMANCE

Intensive development has been carried out on airscrew propulsion efficiency at high forward speeds. Wind tunnel research indicates that propulsive efficiencies between 70 and 75 per cent are attainable up to an aircraft forward Mach No. of 0·8. Blade design to achieve this efficiency level would be transonic for the outer half with thickness : chord gradings of 3 per cent at the tip and around 12 to 15 per cent at the spinner periphery. Blade aerofoil sections now in general use are the N.A.C.A. 16 and 65 series, highly suitable for high forward Mach values. In order to achieve good take-off thrust characteristics there is a general tendency towards the use of higher design lift coefficients which have been shown, both in wind-tunnel and full-scale tests, to give a considerable gain. The degree to which this design lift coefficient can be taken is determined by the level at which a loss in cruise efficiency occurs. Lift coefficients for blades designed purely for optimum cruise efficiency are around 0·4 to 0·5. Recent designs for better take-off are as high as 0 65 to 0·7, although some loss in cruise efficiency becomes evident between these values.

AIRSCREW BLADE MATERIALS

Solid duralumin blades are still widely used on gas turbine power units and, up to 16 ft diameter, will give excellent cruise and take-off performance at an attractive weight for power units up to 4,000 b.h.p. for take-off. In order to prevent any loss in cruising efficiency the electric de-icing elements, instead of protruding above the aerofoil, are being sunk into the blade to a depth equal to their thickness, hence preserving the basic aerofoil shape.

Beyond 16 ft diameter, the hollow steel blade shows considerable weight saving, and is therefore highly desirable at these large diameters. Moreover, the hollow steel blade permits of much wider blade chords without exceptional weight increase, and it is possible for a given power to use three blades in hollow steel as against four blades in solid duralumin, thereby reducing still further the overall weight. The hollow steel blade also facilitates the use of internal de-icing systems which eliminate the necessity for fitting external rubber de-icing shoes which have the effect of reducing the aerodynamic efficiency of the aerofoil. Durability of the blade is also greatly improved by the use of hollow steel construction.

AIRSCREW SYNCHRONIZATION

Development of automatic airscrew-engine synchronization has now been extended to multi-engined turboprop aircraft. Indications are that the automatic synchronizing feature is just as essential on turboprop power units as piston engines, in order to keep all engines running in step for passenger comfort and also to relieve the pilot of the necessity for continual adjustments. By virtue of being an electrical system advantage has been taken to combine electric-head control to the constant speed unit for r.p.m. selection, hence dispensing with long and tortuous mechanical linkage on the larger aircraft.

DUCTED SPINNERS

A development in airscrew design for gas turbines, the value of which has now been proved, is the ducting of the spinner to improve conditions of air entry into the compressor. The conventional central spinner is suspected of 'spilling' the air in front of the intake and may eventually have an outer-shell forming an annulus or duct between this and the present central spinner. The airscrew blade roots in the annulus are carefully streamlined by aerofoil-shaped cuffs which, at the same time, provide the supports for carrying the outer member.

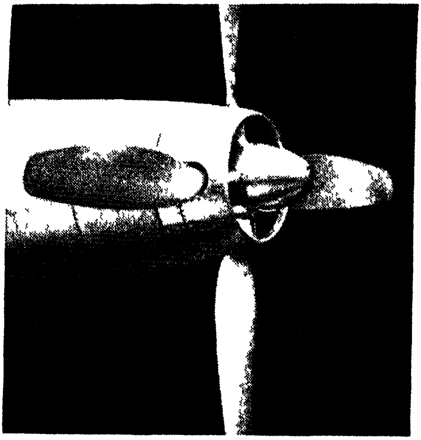


Fig. 4 : 16—Ducted spinner for Napier Naiad.

The weight and complexity of this form of spinner has so far been a deterrent to its general use, accent being given to improve blade root sections with conventional spinners.

SUPERSONIC AIRSCREWS

The main attractions of the supersonic airscrew for multi-engined aircraft is, first, its relatively small diameter, thus permitting undercarriage lengths appropriate to jet aircraft and, secondly, the encouraging data obtained from wind tunnel research on good propulsive efficiencies at forward Mach values of 1·0 to 1·5.

It is questionable, however, whether or not this airscrew will find favour on account of the problem of excessive noise level arising from its high rotational speed. At the moment, there is no solution in sight.

Blades associated with such airscrews will essentially have extremely thin aerofoil sections to realize good propulsive efficiency, the thickness : chord gradings being 2 per cent at the tip and around 6 to 8 per cent at the spinner periphery. To counter the loss in take-off thrust at the small diameter, exceptionally wide chords will be required, extending in parallel form from the spinner to the blade tip and associated also with high design lift coefficients.

Solid-construction duralumin blades are entirely out of the question, first, on account of their lack of torsional stiffness to combat aerodynamic flutter and, secondly, on the grounds of weight. Solid steel or titanium would overcome the torsional stiffness problem, but would be impractical on the grounds of weight, with the consequent high centrifugal forces demanding heavy and cumbersome blade retention. The hollow steel construction offers the only hope of a solution to both problems; developments thus far have been exceptionally promising.

From development work carried out it has become apparent that, using the gas turbine as a power unit, the airscrew still gives the highest overall propulsive efficiency up to a forward Mach No. of 1·0. A further aspect of major importance is that the airscrew will provide some 60 to 100 per cent improvement in take-off thrust over that of a comparable jet engine.

DUCTED FANS

For aircraft speeds of the order of 600 m.p.h. the quantity of air dealt with by an orthodox propeller is very large and the acceleration impressed upon it very small. In other words, jet efficiencies approach 1·0 and losses are mainly profile and compressibility. It becomes reasonable, therefore, to consider the other extreme where, using a much reduced diameter, a smaller quantity of air is dealt with and given a greater acceleration, thereby increasing the jet losses but reducing profile and compressibility losses.

The ducted fan system of propulsion, as we understand it to-day, consists of a fairly large-diameter ducted fan or axial compressor of relatively low compression ratio at the intake to the nacelle. Only a small portion of its delivery air is taken by the turbine, the remainder being by-passed directly to the jet. This system appears to have two main advantages:

(*a*) A slightly higher envelope efficiency than both propeller and jet systems over a relatively small intermediate speed range.

(*b*) It embraces the possibility of burning fuel freely in the air by-passed from the turbine, and thus increasing fairly considerably the power available for short periods, although at an enormous cost in consumption.

However, the disadvantages of the system seem to outweigh these two rather meagre advantages. The main disadvantage is the size of fan and of possible ducts required to deal with the relatively large volume of air required. This would tend to complicate the design of suitable aircraft applications by virtue of the large bulk to be disposed, which might so increase the drag as to nullify any efficiency increase over the jet or propeller system when considering the performance of the aircraft as a whole. There would appear to be no great problems to be overcome from the designer's point of view. Sufficient work has been carried out on similar axial flow fan installations in the past for various applications to give confidence in approaching these designs of the future.

CHAPTER 5

Gas Turbine Components

ROTARY AIR COMPRESSORS are mainly one of two types differentiated by the line of flow in either a radial or axial direction. More aptly termed a centrifugal compressor, the radial type in common use has an impeller with radial vanes running against a smooth casing wall and accelerating the flow of air to a vaned diffuser chamber. The true radial flow type, having alternate co-axial rings of aerofoil section blades arranged parallel to the longitudinal axis on opposing faces of rotor and stator or rotor and counter-rotor, has not appeared in production and would seem to be unsuitable for the high rotational speeds of aircraft turbines. Centrifugal compressors may have one or more stages but the axial type is invariably a multi-stage unit. Some aircraft compressors have several axial stages followed by a terminal centrifugal stage; examples being certain Bristol and Rolls-Royce units.

The compressor on the Heinkel-Hirth jet engine had an initial stage which was not specifically axial or radial, but embodied characteristics of both. The mechanical engineer would probably term this a 'mixed flow' but on aircraft units it is designated a diagonal or semi-axial stage. A further type meriting attention is the Lysholm twin helical-lobe compressor which was included in early Swedish aircraft designs and is employed on large gas turbine stationary power plants in America. Whilst of the rotary type it has the advantage of positive air displacement.

CENTRIFUGAL COMPRESSORS

Due to its long use in industry and as a blower for pressure-charging reciprocating-type aircraft engines, the centrifugal compressor had received many years of intensive development. Performance and reliability were well established so, despite certain inherent limitations, it was adopted for gas turbines. Whilst it could satisfactorily furnish the volume of air required by a piston engine operating with an air: fuel ratio of about 15 : 1 it was less well suited for the turbine-jet unit which requires a fuel: air ratio of from 60 : 1 to 70 : 1. The volume of air to be handled—1,150 ft^3/sec in the de Havilland Ghost unit for example—necessitated an impeller of large diameter and a high speed of rotation, resulting in the linear speed at the periphery being markedly increased. It is not uncommon for the tip speeds of impellers to exceed the speed of sound but in a jet plant it is most desirable to maintain a smoothly accelerating flow and to avoid surging or wave effects.

A centrifugal compressor is necessarily of greater diameter than an equivalent one of axial type and thus the overall diameter of the unit is greater. This constitutes a disadvantage, particularly in the case of wing installation. At the high air speeds of jet-propelled aircraft, a reduction in diameter and consequent lowering of installation drag may well yield an improvement in aircraft performance that will outweigh the advantage of a better specific consumption or a higher thrust. This was exemplified by a comparison of the basically similar German B.M.W. and Junkers plants, both of which had axial compressors. The Junkers 004 developed a thrust of 1,984 lb but the B.M.W. 003, 4 in smaller in diameter and 200 lb less in weight, which had a thrust of only 1,786 lb, gave the same flight speed when installed in the aircraft.

There are, however, compensating advantages when it is compared with the axial type. It is cheaper to produce, more robust, can be run at higher speeds, is less sensitive to dirt and is less prone to icing under unfavourable atmospheric conditions. Furthermore, it has a wider effective operational range than the axial type and in this respect is more suitable for the duty in a variable-speed, variable-load propulsion unit. In a gas turbine, as in all heat engines, design is a compromise between favourable and unfavourable characteristics, and the centrifugal compressor is still successfully employed, for instance, in certain Rolls-Royce and de Havilland units for both combat and transport aircraft.

Air enters the compressor casing by way of the relatively small intake eye around the hub to be picked up by the radial vanes of the rotating impeller, rapidly accelerated and discharged from the periphery into a diffuser. This annular chamber is provided with a number of vanes defining a series of divergent passages, the function of which is to build up pressure in the air stream at the expense of velocity. From the diffuser the air passes to a discharge scroll having one or more outlets (Fig. 5 : 1) or is guided past cascade vanes into ducts leading to the combustion chambers.

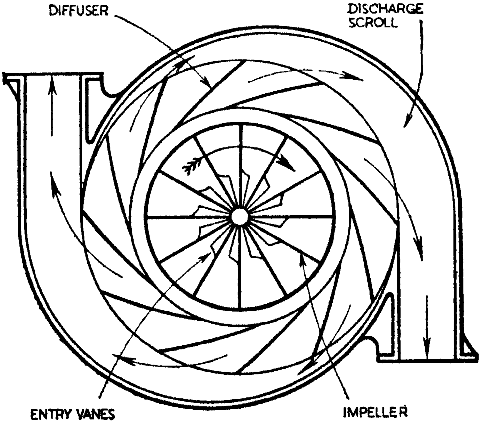
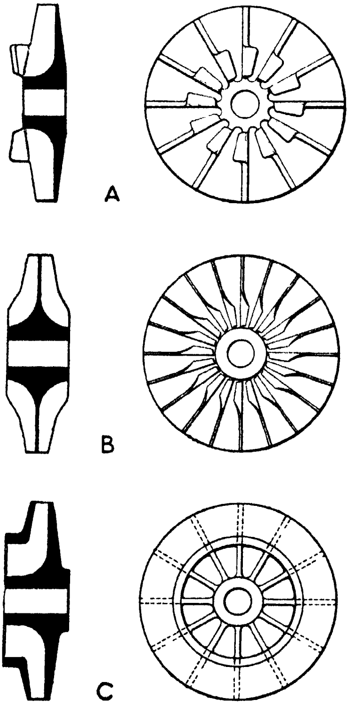


Fig. 5 : 1—Diagram of centrifugal compressor with twin discharge outlets.

Drawings of three different types of impeller are given in Fig. 5 : 2. A simple single-entry ‘ web ’ type A has radial vanes supported on one side by a disc. The radial, divergent passages between the vanes are thus defined on three sides by the impeller and completed by the adjacent casing wall when the impeller is mounted in position. Small curved vanes around the hub are ‘ entry ’ vanes to facilitate the change of flow from the axial to the radial direction and thus reduce the so-called entry shock.

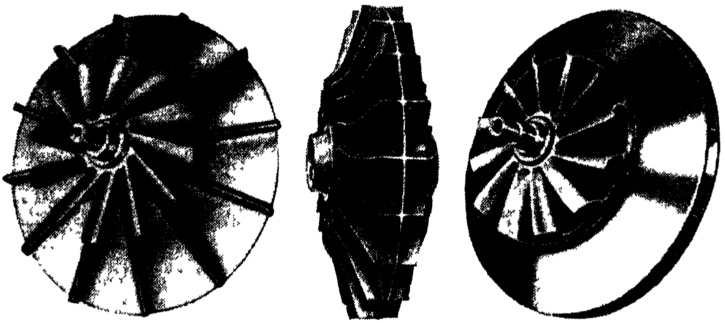
The double-entry type B is a variation of the web impeller with radial vanes on each side of the single central disc. Air enters at each side and is delivered to a common collector duct. It is, in effect, two single-entry impellers arranged back to back. The third impeller C is designated the ‘ closed ’ or ‘ shrouded ’ type. This resembles type A, but has additionally an annular wall on the entry side, and thus the radial passages are completely defined.



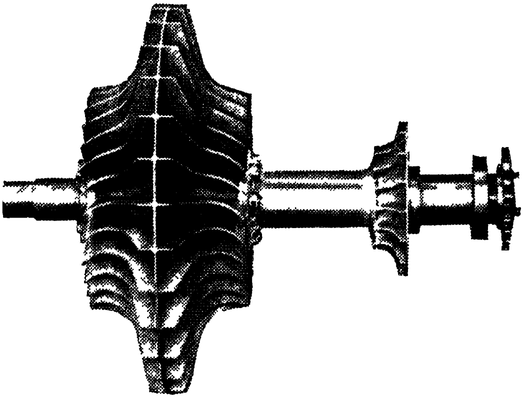
A. Single-entry web type.
B. Double-entry web type.
C. Closed, or shrouded, type.

(Left) Fig. 5 : 2.—Three commonly-used impellers for centrifugal compressors.

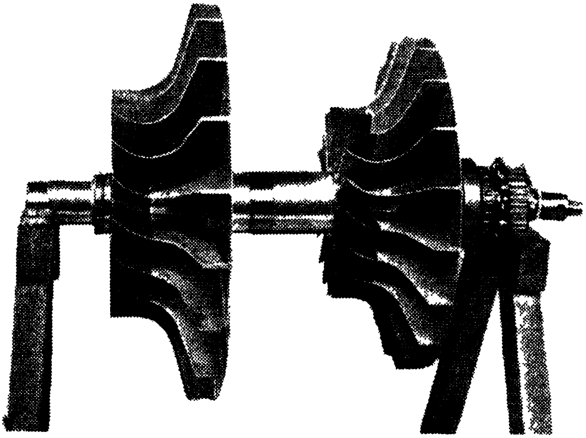
(Below) Fig. 5 : 3—These views of single-entry web, double-entry web and shrouded impellers clearly show the curved entry vanes.



(Right) Fig. 5 · 4—Double-entry impeller of Rolls-Royce Derwent unit. The smaller impeller is for the low-pressure blower supplying cooling air.



(Below) Fig 5 . 5 – Impellers for Rolls-Royce Dart two-stage compressor.



Pictorial representations of the three different types, in the order referred to, are given in Fig. 5 : 3. Actually these show the impellers of turbine-driven supercharging blowers for piston engines There is no essential structural difference for either duty.

The web type A is that in common use as a supercharging blower on piston engines. Type B has the obvious advantage of being subjected to approximately equal forces on both sides of the central web and thus is axially balanced. Examples in this category are the impellers of the Rolls-Royce Derwent, Nene, and Tay units. The fully shrouded type C is more difficult to manufacture, but can usually be designed to produce an efficiency 2 or 3 per cent higher than type A. A variant of this type is the Junkers design with separately boxed passages resembling tubular spokes of a wheel.

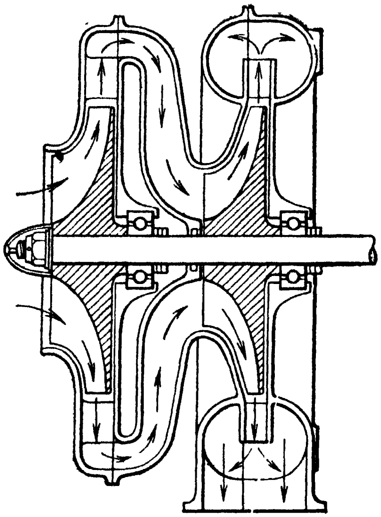


Fig. 5 : 6—Section of two-stage compressor.

It is practical to design centrifugal compressors for a pressure ratio approaching 4 : 1 in a single stage and possible, at some lowering of efficiency, to achieve a higher pressure. Consequently it may be preferable to keep the stage ratio reasonably low and obtain the desired pressures by increasing the number of stages. This substantially increases the axial length of the unit as it necessitates interstage passages to transfer the air from the diffuser of one stage to the intake eye of the next stage, as will be seen in the diagram of a two-stage unit, Fig. 5 : 6. Interstage losses will to some degree lower the overall efficiency.

The pressure ratio per stage increases approximately with the square of the peripheral velocity of the impeller and development in the past has mainly been obtained by raising rotational speed. When the peripheral velocity approaches or exceeds that of sound the mechanical stressing of the impeller is of such magnitude that it becomes almost imperative to arrange the vanes truly radial. Bending stresses in the metal sections are thus avoided and the vanes are stressed solely in tension, but there are concomitant drawbacks. The abrupt change of flow direction renders the air intake difficult and may lead to incomplete filling of the passages between the vanes. Entry vanes are provided on the impeller to ease the change of flow, and as they are located near the hub, the mechanical stressing is not sufficient to preclude the use of curved vanes.

More important, however, is the fact that the velocity of the air leaving a radially vaned impeller is higher than the peripheral velocity of the impeller. Consequently, a practical limit is imposed on the peripheral velocity of the impeller by the absolute velocity of the air which is susceptible, at speeds in excess of the speed of sound, to compressibility factors which reduce efficiency. From low to high rotational speeds within its operating range the efficiency of a typical compressor for a turbojet may fall by 4 per cent. Maximum efficiency attained is about 80 per cent.

Outlet velocity can be reduced by the use of vanes curved backwardly in relation to the direction of rotation, as are employed in pumps and fans, but considerations of strength will enforce a relatively low peripheral velocity. The semi-axial form of impeller is an attempt to circumvent these opposing characteristics. Incidentally, it is known that considerable trouble was encountered on the Heinkel-Hirth design owing to the attached blades coming adrift during operation.

A semi-axial design of some promise by the Swiss firm of Sulzer Frères S.A. is shown diagrammatically in Fig. 5 : 7. The impeller A has integral blades B running obliquely and at an inclination with respect to the axis. Blades are curved from inlet to outlet in a direction opposite to the direction of rotation and the resultant passages between the blades may vary from 30° and 60° to the axis. At least at the outlet edges, the blades in service have a peripheral speed higher than the velocity of sound, but the velocity of the compressed air at the outlet will not exceed the velocity of sound. Consequently it becomes possible to arrange a relatively high ratio of compression in a single stage without suffering a disturbance of flow at the outlet by pressure waves due to supersonic velocity.

The shape of the blades is such that at any plane normal to the axis, say C and D, the cross-section is truly radial, as shown at C_1 and D_1 in dotted outline, in order to withstand centrifugal stressing without being subjected to bending loads. Modifications of this design have additional intermediate blades at the outlet or interrupted and staggered blades, all with the aim of producing a non-turbulent flow of air at uniform pressure.

AXIAL COMPRESSORS

Although of quite distinct character, the axial compressor is subject to the same laws and limitations as the centrifugal type. The line of flow is relatively more direct, but at each stage the air must be accelerated in the direction of rotation and then re-directed to the appropriate angle of attack for the next stage. It is not practical to obtain the same pressure ratio per stage, and an approximate figure of 1 25 : 1 may be noted for the axial type. The trend, therefore, is to employ a relatively large number of stages and maintain the rate of acceleration in individual stages at an

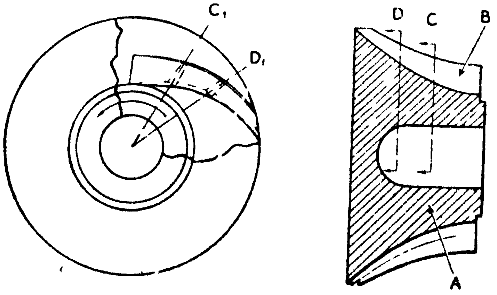


Fig. 5 : 7—Sulzer semi-axial or ‘ diagonal ’ impeller.

efficient level. Regarding each rotor blade as an aerofoil, this is equivalent to avoiding a high lift coefficient and serves to show the intimate relation between mechanical and aerodynamic practice. Actually the axial compressor has been brought rapidly to its present state of practical efficiency by the application of aerodynamic knowledge and by experiment and testing in aerodynamical laboratories.

To transfer the huge volumes of air required for turbine plants in a compressor of reasonable dimensions and weight the blading demands as much care in design as the wing or airscrew blade of a high-speed aircraft. Entry shock must be avoided, Mach numbers must be kept down and every effort made to prevent boundary layer separation and turbulence. As in the normal airscrew blade, the compressor blade must have a varying angle of attack to compensate for the increasing linear velocity from root to tip.

The Germans claimed an efficiency on test of 85 per cent for their axial compressors, but probably 77 to 78 per cent was realized in operation. In modern practice an efficiency of 88 per cent is ~~achieved~~. Delivery volume can be raised by increasing rotational speed and axial velocity but important factors limit development on these lines. Apart from mechanical problems there is the effect on the air stream to be considered. Centrifugal force may upset the stability of the boundary layer at the blades and thus create operational difficulties and even possibly lead to blade stalling. It will be noted that rotational speeds of axial compressors are usually lower than those successfully employed for centrifugal types. An exception is the Westinghouse Yankee unit which rotates at 18,000 r.p.m.

The profile of the annular intake duct to the compressor is of special importance. As in other components some compromise is necessary as it is difficult to reconcile the conflicting characteristics at low and high aircraft speeds. When stationary or taking off there is a tendency for the flow to separate from the inner cone, whilst at high speed the danger of separation occurs at the outer contour. It has been found of advantage to maintain a constant cross-sectional area approximately equal to the area of the compressor intake.

Three stages constitute a probable minimum for an axial compressor, but at the other extreme the number of stages may run to double figures. Of the early German turbojets Junkers had an eight-stage compressor and the B.M.W. a seven-stage unit. The same firms had projected designs with 11- and 12-stage compressors. In Britain an Armstrong-Siddeley unit has a 14-stage axial compressor. Modern developments enable the required pressures to be realized with fewer stages.

Fig. 5 : 8 shows the disposition of parts in a five-stage axial compressor. The casing is usually cylindrical and reduction of the annular working space is achieved by the use of a rotor drum increasing in

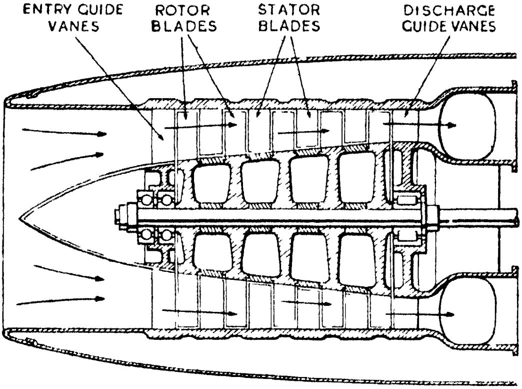


Fig. 5 : 8 — Diagrammatic arrangement of a five-stage axial compressor.

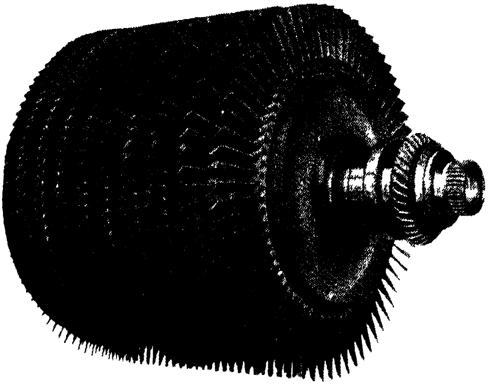


Fig. 5 : 9—Rotor of Napier Eland 10-stage compressor.

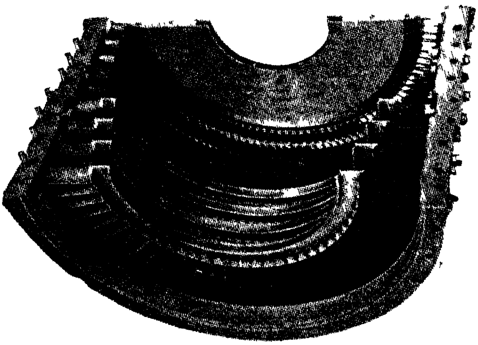


Fig. 5 : 10—Stator half-casing of Eland compressor.

diameter from entry to discharge. Eleven rows of blading are shown as both entry and discharge guide vanes are provided, although discharge vanes and, less commonly, entry vanes may be omitted in certain designs. The rotor is built up of five discs, but in an alternative type discs are employed only at each end and intervening blade rows are carried on a series of rings registering on each other and the end discs to form a hollow rotor drum. A completely assembled rotor and a bladed stator half-casing for the 10-stage Napier Eland are shown in Figs. 5 : 9 and 5 : 10 respectively.

Rotor discs may be of steel or light alloy and will probably decrease in width from low-pressure to high-pressure stages as they accommodate blades of diminishing chord. Steel or light-alloy blades are machined to profile from individual forgings. Methods of fixing the blades to the disc vary in different designs ; they are commonly fitted into tee or dovetail slots in the periphery of the disc and secured by pins, wedges or screws. On the Rolls-Royce Avon the tongued and pinned roots of the compressor rotor blades are solid in the l-p stages and slotted in the h-p stages. The number of blades per disc usually diminishes stage by stage from intake to discharge. Fig. 5 : 12, showing methods of mounting Junkers and B.M.W. compressor blades, is self-explanatory.

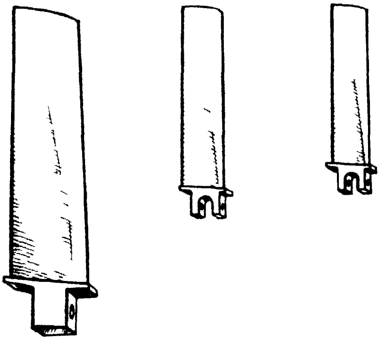


Fig. 5 : 11—Compressor rotor blades for Rolls-Royce Avon turbojet.

Stator blades to direct air flow between stages will also be of either steel or light alloy and probably be assembled in half-rings for mounting in the longitudinally-divided steel or light-alloy casing. They may be machined from forgings, folded from sheet metal, developed from tubes or, in the case of steel, even cambered from a single sheet The casing of the B.M.W. compressor is not divided and the pressed duralumin stator blades are built up in inner and outer rings of magnesium alloy. Both rings are slotted to receive the blades, which are bent and spot-welded to the inner ring, but allowed freedom to expand in the outer ring, as shown in Fig. 5 : 13. The stator rings are loosely positioned on the rotor assembly, inserted in the casing and located by radial set screws.

Earlier reference was made to the relatively narrow operational characteristics of the axial compressor. It is this factor which makes desirable a ‘ bullet ’ to control the effective cross-sectional area of the terminal jet orifice. The blades of an axial compressor tend to stall at low thrust and consequently a bullet, which would probably be a dis-

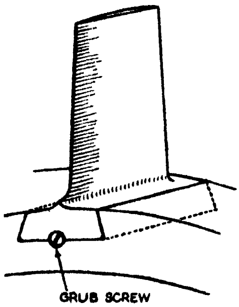
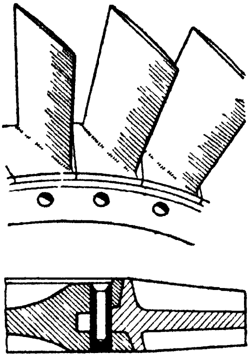


Fig. 5 : 12—Junkers blades located axially by grub screws (left) and B.M.W. duralumin blades pinned in magnesium disc (right).



advantage if used in conjunction with a centrifugal compressor, becomes of practical value. German jet engines employ bullet control devices which, severally, have two positions for idling and power, four positions for starting, take-off, cruising and high speed, and continuous variation determined by the throttle lever and the pitot head pressure.

A too rapid opening of the throttle from idling to maximum position may lead to violent acceleration of rotational speed and stalling of the compressor blades. Should this occur complications may follow rapidly as air delivery falls, air : fuel ratio is changed, combustion conditions become unstable and gas temperature at the turbine becomes excessive. In Germany governed control devices have been fitted, ensuring that the period of acceleration from low to maximum r.p.m. is not less than a predetermined minimum period of several seconds to avoid this contingency and the consequent danger of loss of flying speed and damage to the power unit

Such a limitation, not necessary on a unit embodying a centrifugal compressor, constitutes a handicap in combat performance. To lessen the tendency to stalling by increasing the number of stages and lowering the blade loading, implies an increase in the inertia of the rotor, which militates against rapid acceleration. Doubtless the handicap will be removed as a result of continued development of the axial type.

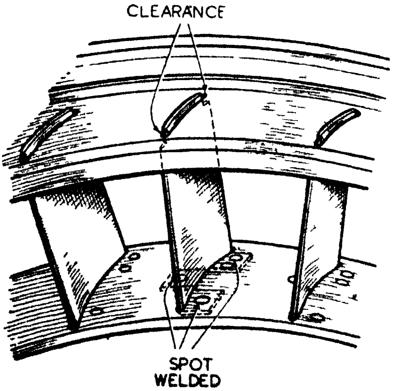


Fig. 5 : 13—Method of fixing stator blades of B.M.W. compressor.

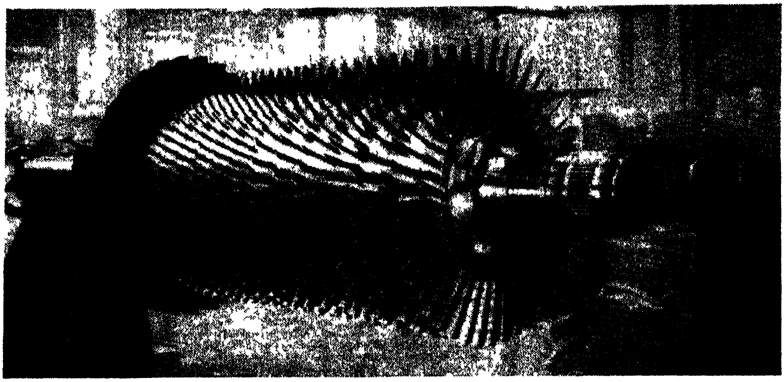


Fig. 5 : 14 —Rotor assembly for Bristol Proteus.

It is useful to compare the two types of turbojet on the basis of thrust related to unit frontal area. Modern examples with centrifugal compressor yield about 450 lb/ft^2 while axial types deliver up to 700 lb/ft^2. For high-performance aircraft, particularly those intended for military duties, the axial type is an inescapable choice.

Reference was made at the beginning of this chapter to compressors having axial stages followed by a centrifugal stage. So far these have been used on turboprops, and in direct-connected and free-turbine types the various stages are combined in a single unit, as in the Bristol Proteus (Fig. 5 : 14). In the case of the compounded-compressor turboprop, axial and centrifugal compressors are independent components driven by independent turbine rotors.

TURBINES

Making use of convenient and not unreasonable approximations, a jet propulsion unit will deliver approximately 50 lb thrust for every pound of air flowing through the unit per second. To drive a modern rotary compressor approximately 100 h.p. is necessary for each pound of air delivered per second. It follows that in a unit developing 2,500 lb thrust the turbine driving the compressor must produce about 5,000 h.p. This estimate is revealing, as it shows an expenditure of energy greater than that of a pair of conventional piston engines of high power output and indicates why performance of a jet-propelled aircraft is so outstanding and why the rate of fuel consumption is relatively heavy. In short, the jet engine is a high-powered unit. Furthermore, it makes clear why the respective component efficiencies and the resulting overall component efficiency are so important.

Dr. Adolf Meyer has stated that an increase in the overall efficiency of the compressor and turbine from 70 per cent to 75 per cent would raise the cycle efficiency from 15 to 18 per cent, representing a 20 per cent improvement. Every 1 per cent increase of component efficiency meant, therefore, an improvement of about 4 per cent in the cycle efficiency.

Turbines may be of the single-stage or multi-stage type but it must be remembered that a loss occurs at each stage and consequently there is advantage to be gained by using only a single stage providing the requisite output can be obtained. This implies high speed of rotation and will tend to confine the use of a single-stage turbine to units employing a centrifugal compressor. At the present level of development a single-stage turbine can provide all the power necessary to drive a single-stage centrifugal compressor. It is lighter, simpler to manufacture and more efficient Multi-stage turbines are more applicable to axial-flow compressors, which operate at lower speeds of rotation, and are also convenient for turbine-driven airscrews where one stage can be used to drive the compressor and the other the airscrew. A single-stage turbine in conjunction with a single-stage centrifugal compressor is the simplest combination for a jet propulsion unit as exemplified in the Rolls-Royce and de Havilland designs of this type. The performance of these British power units in the Meteor and the Vampire aircraft respectively confirm their relative efficiency.

In British practice turbine blades are fully machined from individual forgings of heat-resistant steel or non-ferrous (nickel and chrome) alloys to the closest limits for shape and dimensions and each is subject to what is virtually a laboratory examination. Blades are furnished with profile-

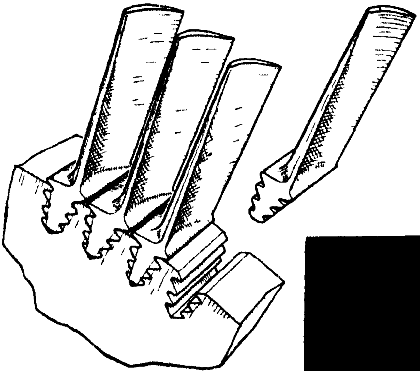


Fig. 5 : 15—' Fir-tree ' blade root fixing on de Havilland turbine.

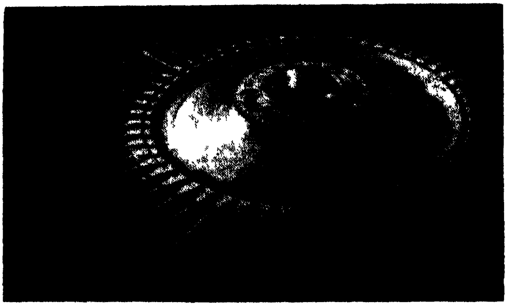


Fig. 5 : 16—Partly assembled turbine rotor for the Pratt and Whitney J.48.

ground ‘ fir-tree ’ roots which are pressed into slots broached in the periphery of the disc and secured by peening on each side, as on the de Havilland Goblin turbine (Fig. 5 : 15). This construction is well shown in Fig. 5 : 16, which illustrates a partially bladed rotor for the Pratt and Whitney J.48—the American-built version of the Rolls-Royce Nene. Produced to close limits of weight, each blade is weighed on a momentum balance, marked to the amount plus or minus of the standard and stored accordingly. Sets of blades are selected, greased, inserted in the previously balanced rotor and peened in a fixture equipped with small compressed-air riveting hammers. After a final operation to align the blades the assembly is degreased and then dynamically balanced. Balance is redressed by the removal of blades and the substitution of other blades having a corrective weight. An example of the use of shrouded blades to minimize gas leakage occurs on the two-stage turbine driving the compressor of the Bristol Proteus 3, shown in Fig. 5 : 17.

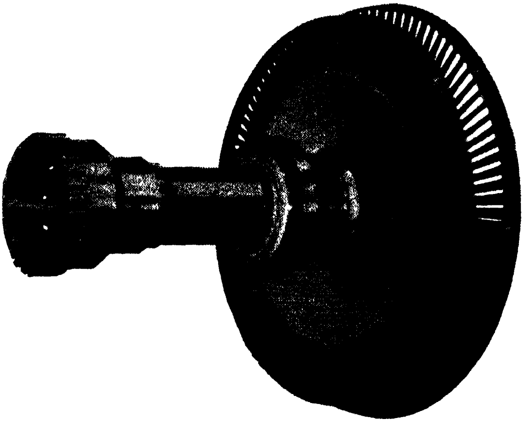


Fig. 5 : 17—Shrouded blades on the two-stage, compressor-turbine rotor of the Bristol Proteus.

For the smaller units turbine rotors may have integral blades, machined on the periphery of the disc. Examples of this method [illegible] in the Rover car turbine and in the Turboméca units.

Similar materials and methods of production are er [illegible] guide vanes of the turbine nozzle. As, however, the vanes are stationary and not subjected to stressing by centrifugal force, they are commonly precision cast by the ‘ lost-wax ’ process. Vanes are fitted in helical slots in the periphery of the inner ring and secured by side clamp rings. On the side adjacent to the rotor the clamp ring may incorporate one half of the labyrinth sealing device between nozzle and turbine blades. As the vanes are continuously exposed to the combustion gases at maximum tempera-

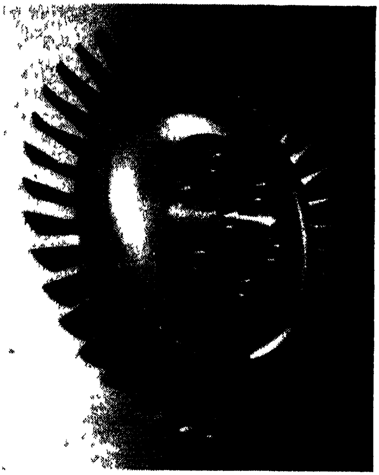


Fig. 5 : 18—Turbine rotor with integral blades in Nimonic alloy for Rover car turbine.

ture they are permitted freedom of expansion in the radial direction and are only guided in the outer ring slots. These features are to be seen in the Derwent nozzle ring assembly (Fig. 5 : 19).

German designers also employed solid blades but, faced with the necessity under war conditions of drastically curtailing the expenditure of alloying material, they developed blades fabricated from sheet or tubular stock with considerable success. Junkers, B.M.W. and Heinkel-Hirth each had different designs of fabricated blades and each had a different method of fixing. For the Junkers 004 the blade was pressed from taper-rolled austenitic steel sheet, folded and welded at the trailing edge. The root, of the thicker section, was formed into a rhomboidal socket and fitted over a projecting stud of the same shape on the periphery of the disc. It was secured by shear pins, as in Fig. 5 : 20, and brazed.

The B.M.W. blade was also of tapered sheet, welded at the trailing

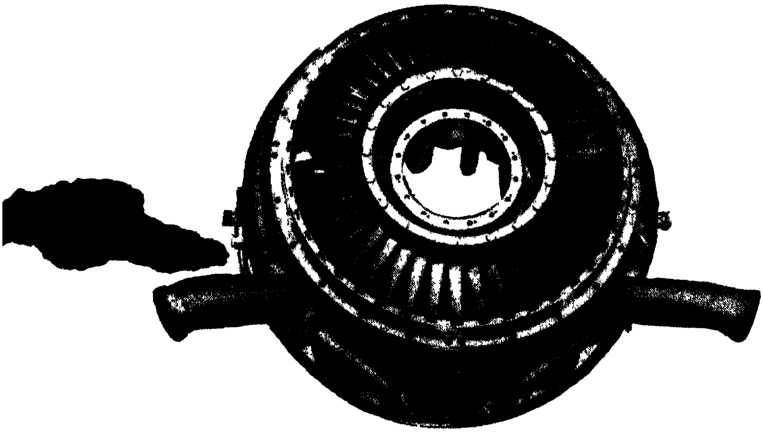


Fig. 5 : 19—Nozzle ring of Derwent turbine, showing radial clearance of vanes.

edge, but furnished with a duplex bulb root which was fitted in a slot in the disc and secured by wedges and pins. Inside the blade was mounted a hollow deflector of similar profile to direct the flow of cooling air along the interior of the blade wall. This air was discharged at the blade tips and joined the main gas flow, so was necessarily at a relatively high pressure. The deflector was retained by a pin fitted in the lower bulb to prevent it collapsing. When slipped into the slot the blade was keyed by two pins at the neck between the bulbs and a pair of triangular-section wedges engaging the upper bulb, as in Fig. 5 : 21. These, it was claimed, tightened on the bulb under the influence of centrifugal action. Side plates positioned the blades axially.

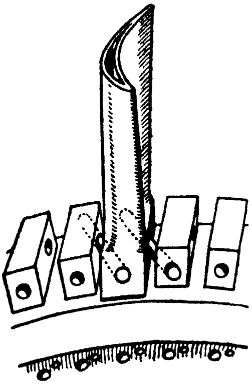


Fig. 5 : 20—Hollow blades of Junkers turbine are pinned and brazed to the disc.

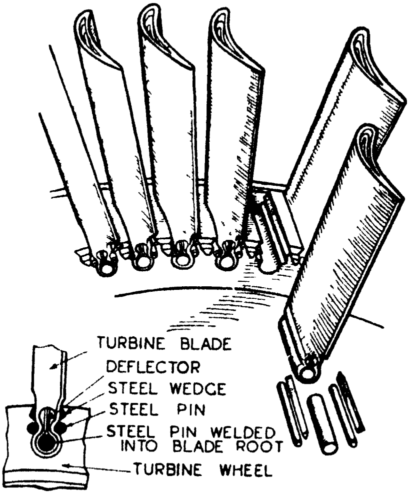


Fig. 5 : 21—Bulb root fixing of B.M.W. hollow turbine blade.

What appeared to be a more attractive design of hollow blade was featured on the Heinkel-Hirth turbine. It was produced by deep-drawing austenitic steel to form a closed-end tube, which was then profiled and shaped at the closed, root end. The root was reduced in width to permit the passage of the anchor pin and furnish apertures for the admission of cooling air to the interior. Stages in the manufacture of the blade are shown in Fig. 5 : 22. When mounted on its anchor pin in the grooved turbine disc, the blade was free to hinge about the pin. Freedom of movement was limited by spacers located between adjacent blades.

Stator blades for the nozzle ring were also hollow and air-cooled in German designs. Usually the cooling air was discharged through slots in the trailing edge. For satisfactory cooling of both stator and rotor

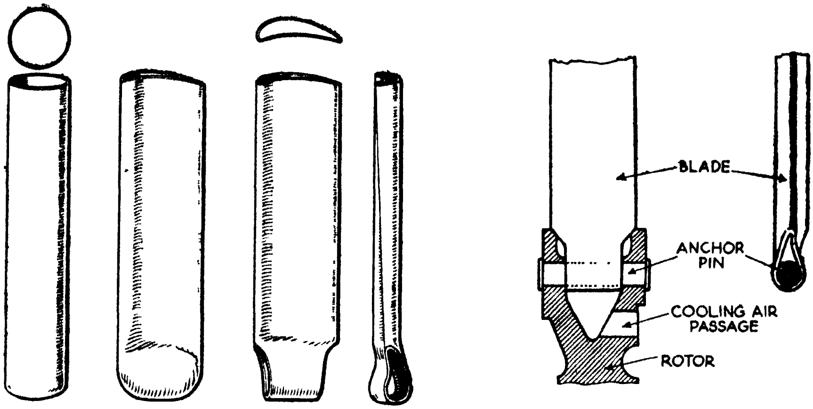


Fig. 5 : 22—Development of Heinkel-Hirth tubular turbine blade and method of mounting on the rotor.

blades, from 5 to 7 per cent of the compressor output may be required, and this constitutes a serious disadvantage.

Under the stress of their war emergency, the Germans investigated a variety of constructions and materials with the aim of making permissible higher gas temperatures at the turbine. A serious effort made to produce a ceramic blade was unsuccessful, due to the low mechanical strength of the material. There would, however, appear to be a reasonable prospect of developing this type for the less highly stressed entry vanes and interstage stator blades.

The Swiss firm of Brown-Boveri has patented a sintered blade which is part metal and part ceramic. By employing the technique of powder metallurgy the material is distributed so that the root of the blade is

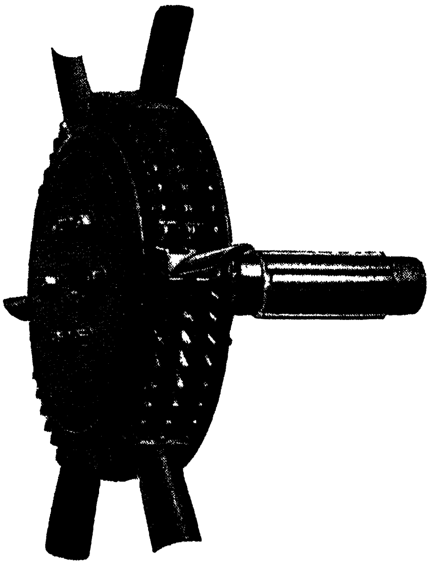


Fig. 5 : 23—Part-assembled laminar turbine rotor developed by Power Jets.

100 per cent metal and the tip 100 per cent ceramic. Details of the practical development of a blade of this type are not yet available. Similar experiments have been carried out in British research establishments.

Problems encountered in the production of the highly-stressed turbine disc are dealt with in Chapter 8. An interesting innovation, patented by J. Atkinson and now under development by Power Jets, Ltd., is the laminar rotor. Discs of Rex 78 austenitic steel sheet are stacked between end checks and drawn together by a series of fitted bolts. As an additional precaution and to relieve the duty of the bolts, the assembly may be brazed up. The periphery of the assembly is then broached to receive the fir-tree roots of the standard blades.

Manufacturing costs should be materially reduced. The ordinary rolling technique in the production of the sheet furnishes the necessary cold working to raise the proof stress, there is lower percentage of waste and inspection is possible at all stages of production. An experimental rotor of approximately 12 in diameter, comprising 15 discs each 0·104 in thick, has been test run successfully in a W.2/700 turbojet for 65 hr at full power and speeds up to 16,750 r.p.m. The possibilities of internal air cooling are being investigated.

CHAPTER 6

Combustion and Fuels

So marked is the tendency to regard the gas turbine as a ' compressor-turbine ' combination that the combustion system commonly receives less attention than is merited. It is fruitless to attempt an assessment of the precise relative importance as all components are essential and functionally interdependent. Nevertheless, it is a truism that the efficiency and performance of a turbine power unit are materially influenced by the combustion system employed. The early German compressors and turbines were not markedly inferior to British components but their less highly developed combustion systems were reflected in lower performance, higher specific fuel consumption and decidedly shorter working life.

Basic requirements for a satisfactory system are a high rate of burning, minimum pressure drop, small bulk and light weight. It must be consistent in operation over a wide range of loads and altitudes, with no liability to flood with fuel or conversely to ' blow out.' Starting must be easy and positive both on the ground and in the air, and combustion must be complete to avoid the formation of carbon.

Entirely new problems were presented by the aircraft gas turbine. For the release of the necessary heat values at the requisite rate of burning there was no precedent. Steam boilers, industrial furnaces or even heat-treatment furnaces are not required to release heat at the rate demanded by the gas turbine and are not handicapped by severe limitations in respect of weight and space occupied.

The pioneer designers were confronted with formidable difficulties as regards combustion systems and received scant sympathy or even attention as it was commonly held that they were demanding the impossible. In 1936 Whittle visited the British Industries Fair at Birmingham and endeavoured to interest manufacturers of oil-burning equipment in the problem. On the ground that his specified requirements could not possibly be achieved he was turned down by all until the firm of Laidlaw, Drew & Co., manufacturers of oil burners for marine boilers, undertook to assist in the design of equipment.

Faced with the same problem, von Ohain in the same year visited the Leipzig Fair and sought the aid of German manufacturers. He was less successful. At that time no one could be convinced that such intensity of combustion was possible.

A solution of the difficult problem of burning weak fuel-air mixtures

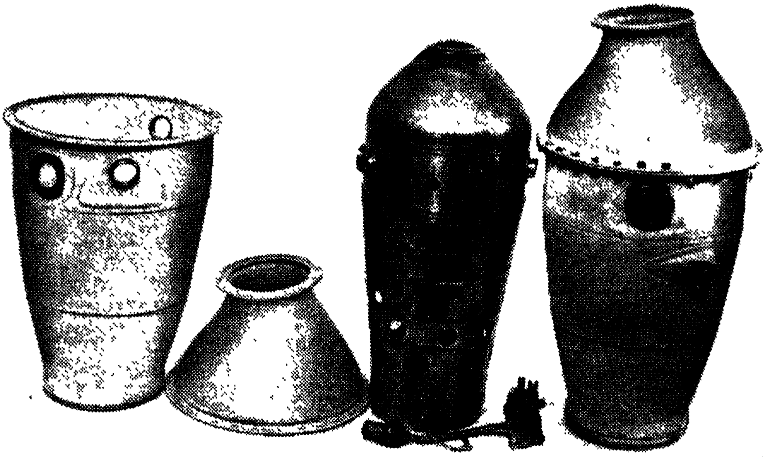


Fig. 6 : 1—Component parts of a combustion chamber. A complete assembly is shown on the right.

in the high-velocity air-streams was found by dividing sufficient air for combustion from the main stream, lowering its velocity and imparting a controlled turbulence. Further air was fed to maintain and complete combustion and the so-called diluent air was added to reduce the temperature of the stream before it reached the entry to the turbine.

In a light tubular combustion chamber of fabricated sheet metal, which by ordinary furnace standards would be regarded as flimsy, and occupying relatively little space, it is already possible to release more than 200,000 B.Th.U/min. To quote one example only of a power unit in regular production will be of interest. The Rolls-Royce Nene has nine combustion chambers each consuming aviation kerosine at the rate of 81·5 gal/hr. The fuel has a calorific value of approximately 150,000 B.Th.U/gal, so each chamber releases more than 200,000 B.Th.U/min. Expressed in another way, the Nene consumes 733·5 gal/hr or 12·2 gal/min. This is an expenditure of potential heat energy at a rate equivalent to 43,000 horse-power.

A combustion turbine may be termed a continuous constant-pressure engine. To enable the turbine to expand the gases to the desired ratio the pressure at the nozzle guide vanes should be as near as possible to the pressure of the air as delivered by the compressor. Fuel cannot be burned, however, unless turbulence is created, and this is only obtained by a drop in pressure in the primary injection and mixing zones. Thus 100 per cent efficiency would not appear to be possible of achievement. Further, as applied to aircraft, the system must function with the minimum loss of unburnt fuel over a wide range of conditions, and thus it is frequently operating below the optimum. Despite the difficulties, combustion systems are operating on standard engines with a total pressure drop of only about 2·5 lb/in^2.

COMBUSTION RESEARCH

Following is a brief description of methods employed at the combustion research laboratories of the Joseph Lucas organization.

In tackling the problem of combustion, four main lines of attack are pursued ; chemical and physical, aerodynamic, thermodynamic, and mechanical. Knowledge of conditions existing inside the flame tube is only possible by the analysis of samples of gas withdrawn from various points. It is of importance to determine how far combustion reactions have progressed at any point in the flame tube, and consequently care must be taken to prevent these reactions continuing whilst the sample is being extracted. To secure such conditions, the sample is withdrawn through a water-cooled tube, the hot gases are rapidly chilled and the reactions brought to a standstill. Subsequent analysis then shows, as nearly as possible, the composition existing at the sampling point at the time of withdrawal. Main gas analyses are carried out on the Haldane apparatus, which permits an accuracy of ± 0·02 per cent. Complications are introduced when samples are taken from the primary combustion zone as it is possible to have unburnt fuel and ‘ cracked ’ gaseous products as well as combustion gases.

Air distribution through the chamber is the subject of detailed study. It is intimately dependent upon flow conditions from the compressor, and particular care is taken in the design of the delivery duct. Pulsation, surging or shock from the compressor delivery would, of course, radically affect combustion conditions. A special traversing apparatus has been developed in the laboratory for investigating the air-

COMBUSTION INTENSITY OF VARIOUS FUEL-BURNING APPLIANCES

Type of Furnace	*Combustion intensity (Therms/ft³/hr/atm)*
Gas burners	
Natural draught	0·5–0/75
Air blast, 2 lb/in^2	2·6–6·2
Concentrated combustion	15
Solid-fuel burners	
Locomotive boiler, full power	5·7
Marine oil-firing burners	
Present-day design, max.	2·9
Gas turbine burners	
1	52
2	49
3	45

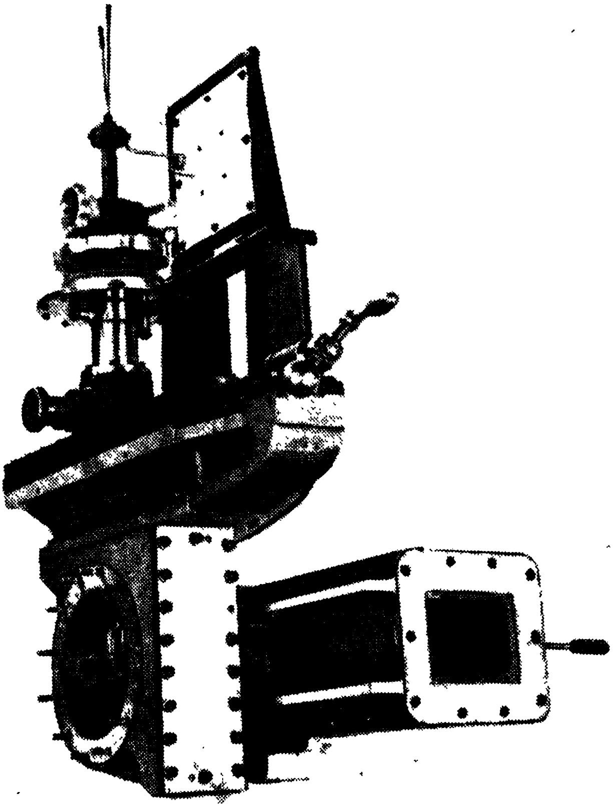


Fig. 6 : 2—Cascade traverse for measuring airflow pattern at the combustion chamber inlet.

flow at the compressor outlet. This enables yawmeter, pitot-static tube and temperature traverses to be carried out at any position in the passage. The first measurement is of yaw in two perpendicular planes. When this is ascertained a pitot tube having the appropriate droop is set up with the correct yaw.

Velocity diagrams obtained by this system of airflow measurement are visually demonstrated by so-called ' pin-cushion ' models. In a base block of shape and area corresponding to the section of the delivery passage, wires of a scale length to show velocity are set at the appropriate angle to indicate flow direction. In the case of a reverse flow in turbulent conditions this is represented by a loop. Optical methods of investigating airflow are also employed. Flow lines are observed in a cascade box fitted with Perspex sides by means of a Schlieren set-up and are recorded

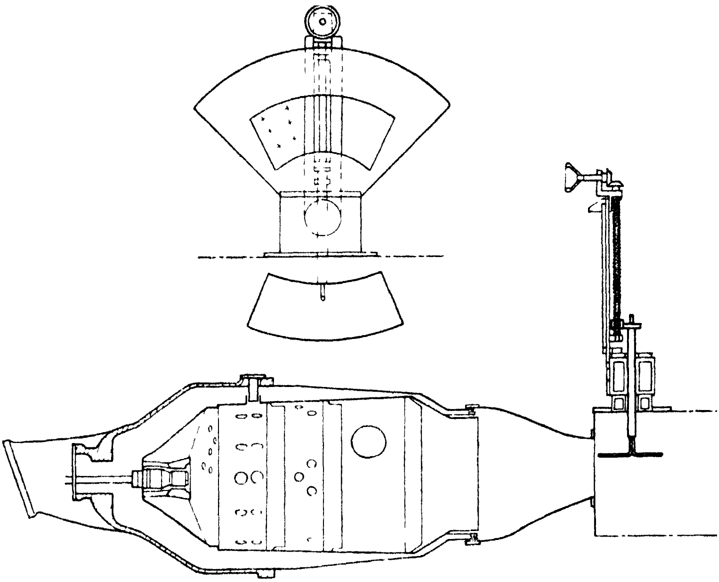


Fig. 6 : 3—Typical exhaust traverse arrangement.

with an ultra-high-speed camera. Similar traversing equipment is employed to determine pressure, velocity and temperature at the opposite end of the chamber, where the gases are delivered to the nozzle guide vanes of the turbine.

Contour diagrams of temperature and velocity at the segmental entry to the turbine are given in Figs. 6 : 4 and 6 : 5 respectively. The aim is to obtain a uniform temperature distribution, but some latitude can be permitted if the highest temperature does not occur in the vicinity of the turbine blade roots where the metal is most highly stressed.

Heat transfer from the gas to the metal is at a high rate, and the heat-resistant alloys used for blades have a relatively low rate of thermal

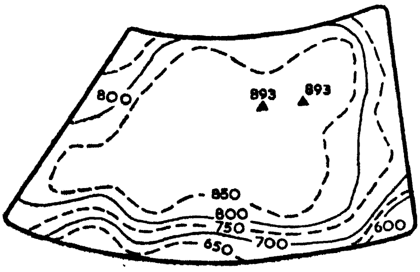


Fig. 6 : 4—Temperature contour diagram (°C)

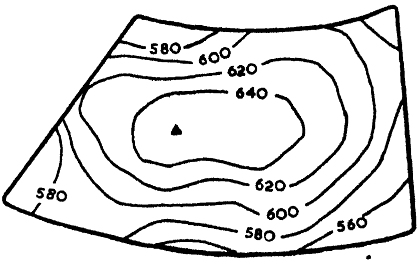


Fig. 6 : 5—Velocity contour diagram (ft/sec)

conductivity. Consequently, heat flow along the blade can do little to remedy any irregularity in temperature distribution.

Combustion efficiency may be defined as :

$$\frac{\text{Actual Temperature Rise}}{\text{Theoretical Temperature Rise}}$$

It will be appreciated that extremely careful measurements of air supply, fuel supply and assessment of calorific value are necessary to obtain an accurate estimate of the theoretical temperature rise. Traversing instruments carrying thermo-couples are used to determine the actual mean temperature at both the inlet and outlet of the chamber. At the discharge end the gases have a mean temperature of about 800° C and a velocity of the order of 500–600 ft/sec. It is, of course, essential that combustion be complete before reaching the turbine nozzle ring. Flame, unburnt fuel, or a stratified flow at different temperatures would have a most detrimental effect on the turbine blading.

A complete power unit may have a single annular combustion chamber or a plurality of individual chambers varying from two to 16 in number. The trend has been, however, towards fewer chambers, as exemplified by the change from 16 on the Goblin to ten on the Ghost, and from ten on the early Derwents to nine on the Nene and Derwent 5. The outer casings, joined by interconnecting pipes, were commonly tapered from the entry end to the manifold which receives the hot gas and conducts it to the turbine nozzle. Latterly, with the more compact axial-flow power units, the casings tend to be cylindrical. Inside each casing is arranged a flame tube, and these are again connected by balance pipes located within the casing interconnectors. These serve as pressure equalizers and also enable flame to be propagated around the complete assembly from an initial ignition point in one chamber.

Flame tubes are of Inconel or Nimonic nickel-chrome alloys, whilst

CHARACTERISTICS OF TYPICAL CHAMBERS

Power Unit	1	2	3	4	5	6	7	8
Mean air velocity at entry (ft/sec) ..	178	206	359	343	364	240	387	212
Mean kinetic head at entry— (per cent of total pressure) ..	0·99	1·38	4·33	3·87	4·31	1·72	4·54	1·38
L/D ratio = $\frac{\text{Length of gas path}}{\text{Outer casing dia.}}$ ..	4·4	3·91	2·70	2·45	2·26	2·55	2·34	3·12
Net loss of total pressure (per cent)..	3·31	3·90	3·64	6·95	7·52	3·0	7·5	2·7
Loss of total pressure of elbow or inlet passage (per cent)	2·05	2·85	3·92	4·25	3·99	1·32	1·7	1·4
Loading ratio = $\frac{\text{Fuel flow per chamber}}{\text{Weight of chamber}}$ $\left(\frac{\text{lb/hr}}{\text{lb}}\right)$	7·9	11·3	9·75	17·05	21·3	8·9	—	5·65
Specific thrust = $\frac{\text{Thrust}}{\text{Power unit weight}}$ $\left(\frac{\text{lb}}{\text{lb}}\right)$	1·837	1·955	2·03	—	3·4	—	—	—
Combustion intensity (Therms/ft^3 hr/atm)	29·8	43·7	32·9	51·6	49·1	34·9	33·5	41·4

PERFORMANCE OF TYPICAL STRAIGHT-THROUGH CHAMBERS

Power Unit	*No. of Chambers*	*Fuel consumption gal/hr*	*Air Mass Flow lb/sec*	*Air : fuel ratio*
Experimental	6	17·2	2·3	60 : 1
Derwent V.	9	57·0	7·0	55 : 1
Ghost	10	67·0	8·8	59 : 1
Nene	9	81·5	10·0	55 : 1

the outer casings are usually of mild steel. Apart from obtaining satisfactory combustion performance it is obviously most important to extend the reliable working life of the components. In new types a working life of 300 hr for flame tubes is usually acceptable but the routine service life in fully developed engines is 500–600 hr and tests have shown that they can exceed these figures in operation.

In the early days it was customary to divide the combustion chamber into a primary zone adjacent to the burner, a secondary zone where air was added to complete combustion and a tertiary zone where air was introduced to dilute the stream of gases and lower the temperature. As an atomized fuel spray is used, however, combustion is more akin to that in a diesel engine than in a gas furnace. Hence the primary zone might be described more accurately as the combustion zone and the volume further downstream in the flame tube as the mixing zone.

The operation of the combustion system is intimately associated with the airflow pattern from the compressor, and with centrifugal compressors ideal conditions are far from attainable. The air stream received in the diffuser channels from the impeller is not uniform and has to be turned through approximately a right-angle. Not only is the velocity of the entering air high, but the motion over the cross-section of the stream is far from uniform either in velocity or direction.

Some 75 per cent of the total airflow enters the annulus between flame tube and outer casing, and after cooling the upper portion of the flame tube surrounding the combustion zone passes through the dilution holes. About 25 per cent passes directly into the combustion zone through various devices which have the object of imparting to this portion a definite and controlled pattern.

Quite early Metropolitan-Vickers (now Armstrong Siddeley) and contemporary American designs incorporated an annular combustion chamber. This type was a feature of B.M.W. and Heinkel-Hirth in Germany. Inner and outer components of the B.M.W. chamber with the characteristic air mixing slots are shown in Fig. 6 : 6, whilst Fig. 6 : 7 is a section of the assembled chamber and indicates the airflow path.

To-day nearly all new designs of aircraft gas turbine incorporate a

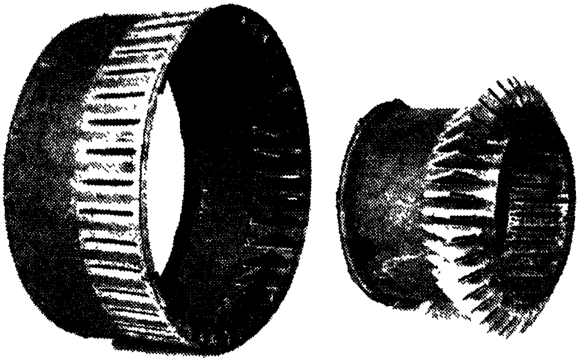


Fig. 6 : 6—Main components of B.M.W. 003 annular combustion chamber.

combustion system either of the annular or cannular type. In the former (Fig. 6 : 8) combustion takes place within the annular space between inner and outer circumferential flame tubes. The cannular system incorporates individual flame tubes spaced in an annular combustion space; this has most of the merits of the true annular system and is also somewhat easier to develop on test rigs. The combustion chamber of the Pratt and Whitney J57 (Fig. 6 : 9) is an excellent example of a modern cannular design.

FUEL CHARACTERISTICS

From the earliest days of the aircraft gas turbine, kerosine has been regarded as the most suitable fuel. It commends itself on the score of availability, cost, calorific value, combustion characteristics and a low vapour pressure giving reduced fire hazard. Availability is a prime consideration, both in time of peace and, to an even greater extent, under war conditions. It involves not merely the quantity available, actually or potentially, at the refineries but also transport, distribution, storage and handling facilities. It was such considerations, the so-called logistical reasons, that led to the more widespread use of motor gasoline for gas

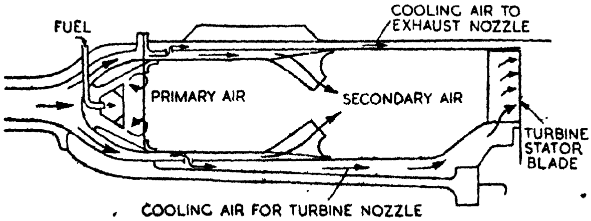


Fig. 6 : 7—Diagrammatic section of B.M.W. annular chamber showing flow path.

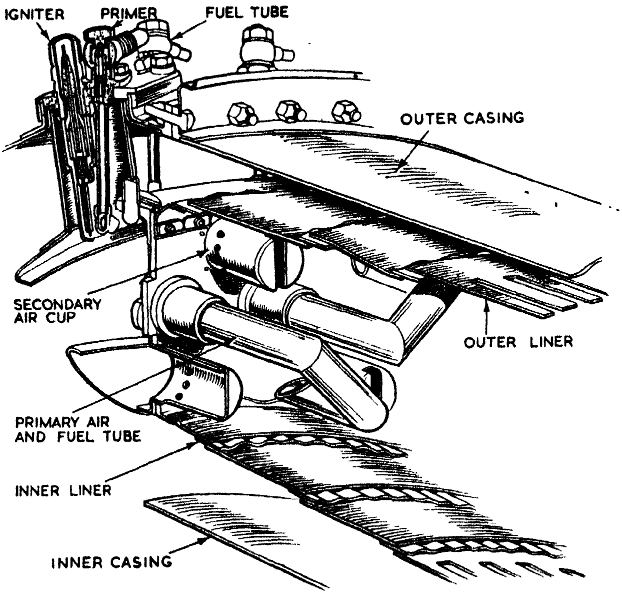


Fig. 6 : 8—Section of annular chamber on YJ65 Wright Sapphire.

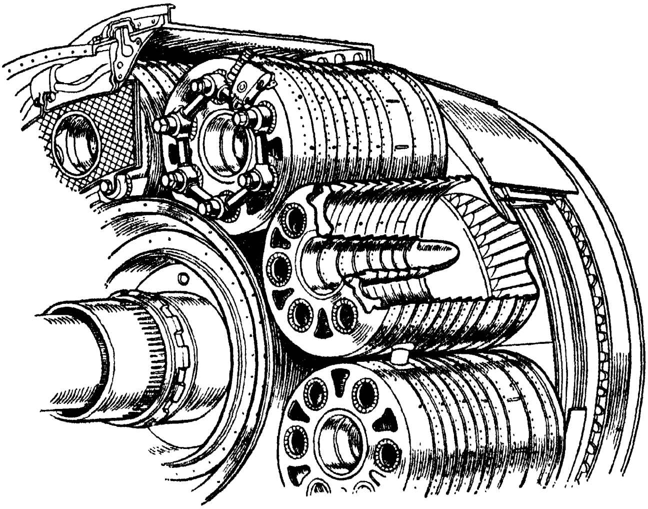


Fig. 6 : 9—Cannular chamber on Pratt and Whitney J57 turbojet.

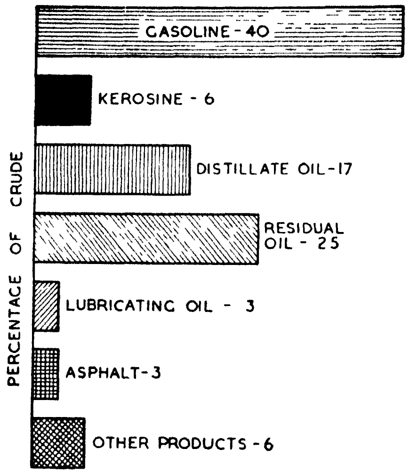


Fig. 6 : 10 — Yield of products from petroleum crude.

turbines in the U.S. Forces. Due to the ubiquitous use of kerosine over a long period for heating and lighting, all these facilities and the necessary experience exist. As regards quantity of supplies, this is regarded as adequate for civil aviation in the foreseeable future, despite the fact that the yield of kerosine from the petroleum crude is relatively low.

Due to the exceptionally high rate of heat release required by the gas turbine unit, a special importance attaches to the calorific value of the fuel used. In general, the calorific values of petroleum fuels are related to their specific gravities. With a heavier fuel there is a higher calorific value by volume but a lower calorific value by weight. Thus the significance of the calorific value will depend upon the aircraft application. In modern, high-performance, military aircraft of the fighter or interceptor types, tankage capacity is likely to be a limiting factor and a heavier fuel would be desirable at operational ranges below that at which the fuel weight becomes critical. For both military and civil transports, however, in which the decisive factor is normally the ratio of fuel weight to payload, a lighter fuel having a higher calorific value per unit weight could be more advantageous.

The accompanying table shows the specific gravities of various fuels and their calorific values on both a volumetric and a weight basis. The reference fuel, RDE/F/KER, is the British " Pool burning oil " which during and immediately after the war served as the standard aviation kerosine and is now displaced by kerosine to the specification DERD 2482. This does not limit specific gravity but calls for calorific value 18,300 B.Th.U./lb minimum, flash point 38° C minimum, freezing point –40° C maximum, and viscosity 6 cs at – 17·7° C.

To cover possible emergency in time of war, both Britain and U.S.A. have laid down specifications, DERD 2486 and JP–4 respectively, for wide-cut distillate fuels that embody portions of the gasoline, kerosine and gas oil ranges and effect a compromise between combustion requirements and availability. Being intended for military use they naturally

CALORIFIC VALUES OF VARIOUS FUELS

Fuel	*Specific gravity*	*Calorific Value (net)*		*Calorific Value (ratio to RDE/F/KER)*	
		B.Th.U/lb	*B.Th.U/gal*	*per lb*	*Per gal*
100/130 grade aviation gasoline	0·720	18,940	136,580	1·0204	0·9081
Motor gasoline (average)	0·740	18,890	139,900	1·0177	0·9302
JP-4 (Wide-cut distillate)	0·739 0·825	18,700	143,500	1·0075	0·9541
DERD 2482 (British aviation kerosine)	0·780 0·810	18,600 18,700	146,000 151,000	1·0048	0·9874
RDE/F/KER (British Pool burning oil)	0·810	18,560	150,390	1	1
Diesel oil (British road vehicles)	0·848	18,710	158,800	1·0080	1·0559
Gas oil	0·850	18,260	155,230	0·9838	1·0321
Light diesel oil	0·870	18,050	157,040	0·9724	1·0442
Heavy diesel oil	0·910	17,980	163,630	0·9687	1·0881
Tetralin (Tetra-hydro-naphthalene)	0·970	17,400	168,800	0·9375	1·1224
Boiler fuel	0·973	17,800	173,160	0·9590	1·1514
Cracked fuel oil (California)	1·013	16,910	171,300	0·9110	1·1390
Cracked fuel oil (Texas)	1·080	16,110	174,000	0·8679	1·1569

stipulate a low freezing point, —60° C maximum, and pumpability at low temperatures to permit cold starting and operation at extreme altitudes.

Modern aircraft realize their high performance economically only at altitudes of 40,000 to 50,000 ft. At such heights the ambient air temperature is commonly regarded as being fairly constant at about —55° C but it may fall to —80° C, particularly over tropical areas. Of course, with the rapid rate of climb currently achieved, the temperature of the fuel will not fall at the same rate as the air temperature and consequently, on short- or medium-range operations, may never be reduced to that of the ambient air. Much will depend upon structural considerations. With integral wing tanks the drop may be 20° C in the first hour of operation while with bag tanks the rate may be as low as 6° C per hour. If long-range civil aircraft are to cruise at 45,000 ft or over, however, kerosine will barely suffice unless fuel warming or, alternatively, tank insulation is provided.

Although wide-cut fuels will be acceptable for military duties on the score of availability, they will not be welcomed for civil operation on account of their increased volatility. Troubles in the fuel system arising from vapour evolution can be more severe on turbine-powered aircraft than on piston-engined aircraft due to the higher rate of climb and consequent less time for fuel cooling. Precautionary measures will need to be taken to ensure adequate vapour capacity and to avoid the possibility of choking the pumps.

Associated with this problem is the potential loss of fuel by evaporation, unless the tanks are pressurized. It has been estimated that if aviation gasoline is taken rapidly to 40,000 ft altitude the loss would be 7 per cent if the initial temperature was 29° C and 12 per cent if the temperature was 43° C. Actually, these figures may be exceeded as some liquid fuel may be carried over by the vapour. This is a serious factor for the civil operator since, as the take-off weight cannot be exceeded, it implies a reduction of range or payload. Approximate figures for the Reid vapour pressure at 38° C of typical fuels are :

Aviation gasoline	7 lb/in^2 abs
Wide-cut distillate	2–3 „ „
Kerosine	0·1 „ „
Gas oil	0·03 „ „

Probably the major deterrent to the use of high vapour pressure fuels, particularly in the case of civil operators, is the increased fire risk in the event of a crash landing. There are other hazards in the air, such as the self-ignition of fuel leaking on to a heated surface or the ignition of fuel-air vapour in a tank, but their incidence is lower and the crash fire is regarded as the most serious.

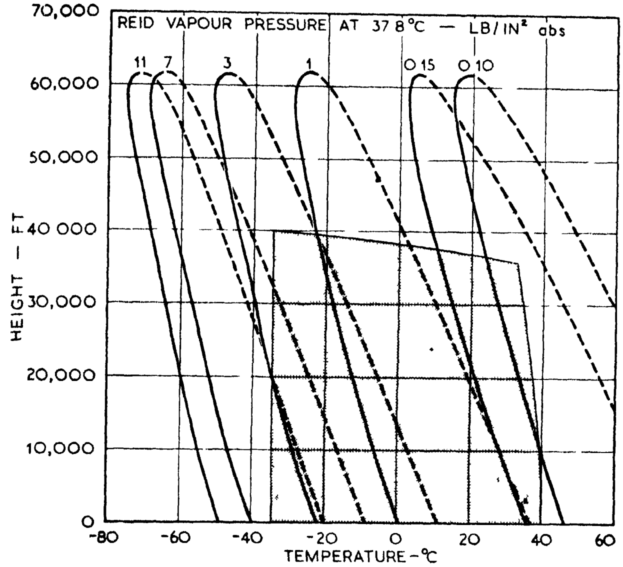
No fuel is completely safe. The flash point, or lower limit of inflammability, of a fuel is reached at a vapour pressure of about 0·155 lb/in^2 abs, which is attained at quite low temperatures for the more volatile fuels, approximately as shown by the curves in Fig. 6 : 11. At ground level the approximate minimum temperatures to which fuels must be heated to produce inflammable mixtures are:

Aviation gasoline	—40° C
Wide-cut distillate (DERD 2486)	—20° C
Aviation kerosine	35° C
Gas oil	65° C

There exists the possibility, of course, that a small fire initiated by burning oil or an electric spark may operate as a pilot flame and progressively heat up the main body of the fuel. In this respect the heavier fuels show to advantage as relatively more time must elapse before the pilot fire can heat up the fuel to the flash point at which a catastrophic fire ensues. The wide difference in the critical temperatures of kerosine and either wide-cut distillate or gasoline is taken to indicate a substantial reduction of the fire hazard.

As regards the tendency to self-ignite on contact with a hot surface the position is reversed, the temperature varying in approximately inverse ratio to the density of the fuel. Kerosine has a lower spontaneous ignition temperature than gasoline, about 650° C against 725° C, but lubricating oils ignite at much lower temperatures, say 355° C to 450° C, and thus present a greater hazard. It is probable that in many crash fires, lubricating oils or hydraulic fluids are responsible for the initiation of the conflagration.

All fuels hold air in solution which is released when the aircraft climbs to its cruising altitude. Some criticism has been directed towards kerosine on the grounds that it can produce an explosive mixture in the



——————— lower limit   — — — — — upper limit

Shaded area shows fuel temperature range measured on Comet

Fig. 6 : 11 —Limits of inflammability of hydrocarbon fuels at various temperatures and altitudes.

tank at temperatures ranging from 10° C to 80° C and at altitudes up to 40,000 ft. Under such conditions gasoline produces a mixture too rich to come within the explosive range. The relative risk depends in the main upon the ambient temperature. Under tropical conditions the risk is higher with kerosine than with petrol, under temperate conditions the hazard is about equal, but at low temperatures, 0° C and lower, kerosine is quite safe while with gasoline the risk remains considerable.

On the graph of Fig. 6 : 11 is superimposed the operational fuel temperature on the Comet turbojet liner, as quoted by Mr. J. E. Walker, Chief Power Plant Engineer of the de Havilland Aircraft Co., in a lecture given before the Royal Aeronautical Society. It will be noted that with wide-cut distillate (Reid vapour pressure 2–3 lb/in^2 abs) explosive mixtures will occur over a range of temperatures and altitudes which make it more hazardous than kerosine. For civil aircraft the danger can be averted by appropriate design measures ensuring there is no possible source of ignition in the tank and by adequately venting the tank. In the case of military aircraft exposed to enemy gunfire the fuel in the tank may be blanketed by an inert gas such as nitrogen.

As kerosine evaporates more slowly than gasoline it has been suggested that, in the event of a leak in the tank or plumbing, it could accumulate in the wing or fuselage and there be a source of vapour formation. In such circumstances, however, any fuel could be a potential danger. Design, inspection and maintenance should reduce the risk of leakage to a negligible minimum and the accidental leakage should be provided for by good drainage and ventilation of the structure.

Clean fuel is essential for satisfactory operation in view of the fine working clearance of the injection components and the relatively restricted passages in the burners. In Britain the tanker vehicles employed for refuelling jet aircraft are equipped with Streamline filters to prevent the transfer of foreign matter to the aircraft tanks. As kerosine has virtually no lubricating properties and, in fact, has a scouring action, it is customary to make an addition of 1 per cent of lubricating oil to the fuel to ensure satisfactory operation of the injection system. The oil also serves as a corrosion inhibitor.

CHAPTER 7

Fuel Systems

In view of the vast quantity of air flowing through a gas turbine, the high velocity of the flow, and the critical character of the gas temperature ahead of the turbine, the feeding of fuel presents a problem of considerable magnitude. It is difficult enough in the case of a stationary unit operating under an approximately constant load in relatively stable conditions. For an aircraft propulsion unit which must operate under varying load in conditions of wide and rapid concurrent changes of air temperature, density and pressure, the problem becomes prodigious. As might be expected, finality in regard to fuel systems and equipment has not been attained. Four systems used on turbines in regular production may be differentiated according to the type of fuel burner employed ; whether of the high-pressure atomizing, low-pressure spill, vaporizing or rotating-jet types. There is available a wide range of components employed for the injection, automatic regulation, and control of fuel supply, and description of the construction and operation of a number of these follows The trend is to standardize component items of equipment and to " tailor " the system to the requirements of the power unit in specific installations.

LUCAS PRESSURE ATOMIZING SYSTEM

In Fig. 7 : 1 is given a diagrammatic layout of a typical installation ; that for the Rolls-Royce Nene turbojet, developed by Joseph Lucas Ltd in collaboration with the engine manufacturer. The provision of two high-pressure pumps in parallel is merely to obtain the requisite high rate of delivery from standardized units and involves no operating principle. It should be noted that when dual pumps are fitted, present-day practice is to incorporate a solenoid-operated isolating valve as a safeguard against the failure of one of the pumps when taking-off or landing. The pumps shown in the diagram are of an early type with the cam plate mounted on a ball-bearing in the control ring.

FUEL PUMP

The seven plungers of this positive-displacement pump reciprocate in a rotor running in two carbon bushes and driven from the engine by a splined quill shaft. The bores for the plungers are spaced evenly around a pitch cone concentric with the rotor axis and converge towards the inner or pumping end. Reciprocation of the plungers is effected by

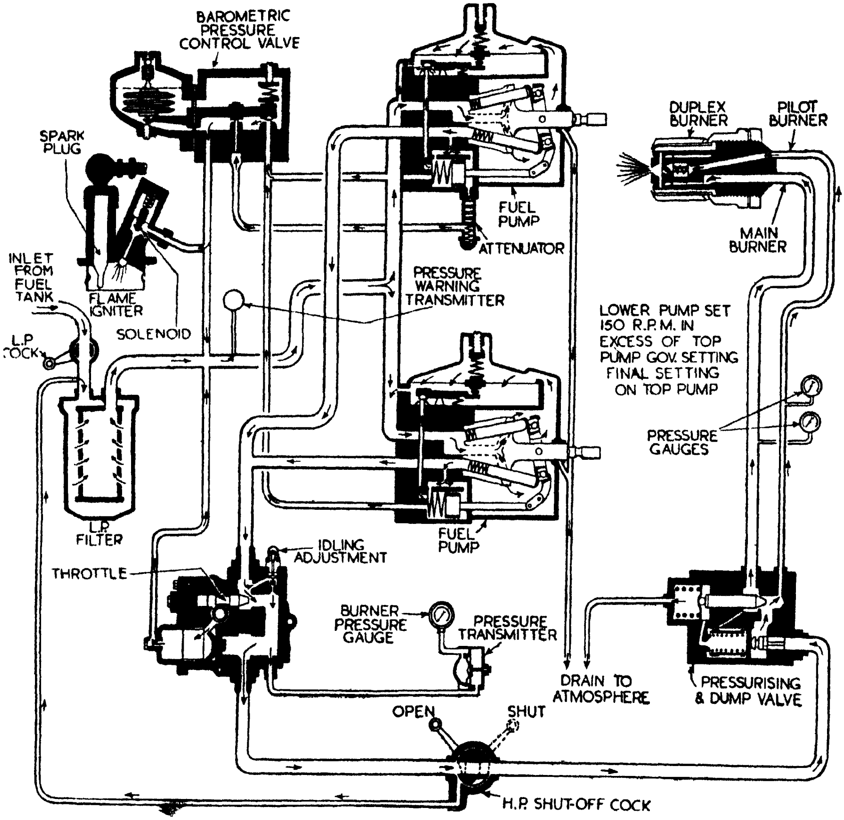
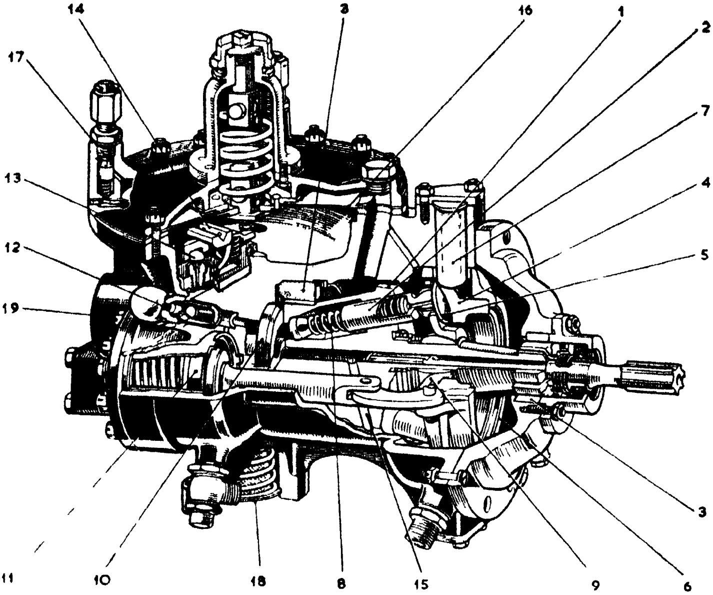


Fig. 7 : 1—Diagrammatic layout of Nene fuel system.

means of a cam plate engaging the outer ends of the plungers which are ball-shaped to take socketed slipper heads. The cam plate is carried in a control ring swivelling on a pair of trunnion pins set at right angles to the rotor axis. On present pump designs the angle of the cam plate relative to the rotor axis may be varied by 15°. The stroke of the plungers at maximum delivery varies with the size of the pump. On the one fitted to most current turbine units the stroke is approximately 0·706 in. Retraction of the plungers on the inlet stroke is by helical springs located against the reduced inner ends of the bores.

As the pump is designed to handle aviation gasoline or wide-cut distillate as well as kerosine, special care has been taken to ensure smooth running despite the non-lubricating nature of the fuels. A small-diameter axial hole in the plunger ball-end admits a cooling film of fuel to the



1 Rotor 2 Plunger 3 Carbon bush 4 Cam plate 5 Slipper 6 Control ring 7 Trunnion pin 8 Return spring 9 Retaining plate 10 Valve insert 11 Pressure control piston 12 Restricting orifice 13 Plate valve 14 Rocker 15 Centrifugal drillings 16 Diaphragm 17 Bleed valve 18. Intake 19 Delivery

Fig 7 2—Arrangement of Lucas ' C ' size high-pressure fuel pump

underside of the slipper and another hole in the slipper permits leakage of fuel to the slipper working face. To obviate any possibility of the plungers sticking in their bores, a drilled retaining plate, running on a spring-biased hemispherical bush, holds the slippers up to the cam plate.

At its inner end the face of the rotor is ground flat and smooth to make a pressure-tight seal with a valve insert furnished with two kidney-shaped ports communicating respectively with the pump inlet and the pump delivery.

Parallel with the rotor axis is a bore in which the pressure-control piston operates. The piston rod is connected by a link to a lug on the control ring, and concentric helical springs bias the piston to move the control ring to the position for maximum stroke of the pump and consequently maximum output. From the delivery side of the pump, fuel under pressure is applied below the piston in opposition to the springs and also, by way of an adjustable restricting orifice, above the piston. From this space above the piston, fuel can escape by way of a plate valve which is normally held to its seating by a spring-loaded rocker. By this means the fluid forces acting on the piston are balanced and there is no resulting movement. Should, however, the fluid pressure exceed a value determined by the spring loading of the rocker, the plate valve will be lifted from its seating and the flow through causes a pressure drop to occur across the restricting orifice and creates a state of unbalance. Under this influence the piston moves to a new position, reducing the stroke of the plungers and consequently the output of the pump.

Incorporated in the pump is an overspeed governing device, the purpose of which is to limit the speed of the turbine to a predetermined maximum by automatically controlling the fuel delivery from the pump. In the rotor are seven drillings extending from the maximum peripheral diameter to an axial bore which is in free communication with the suction chamber of the pump. In operation, centrifugal force produces in the drillings a pressure difference which results in the fuel in the pump casing surrounding the rotor and also in the space above the diaphragm being maintained at a pressure higher than obtains in the suction chamber. This pressure difference is utilized to move the diaphragm, against the constraint of a helical spring in tension, so that it depresses the rocker and unloads the plate valve. The pump output is then reduced in the manner already described. The spring loading of the diaphragm is readily adjustable so that the engine speed at which the governing action begins may be set with accuracy.

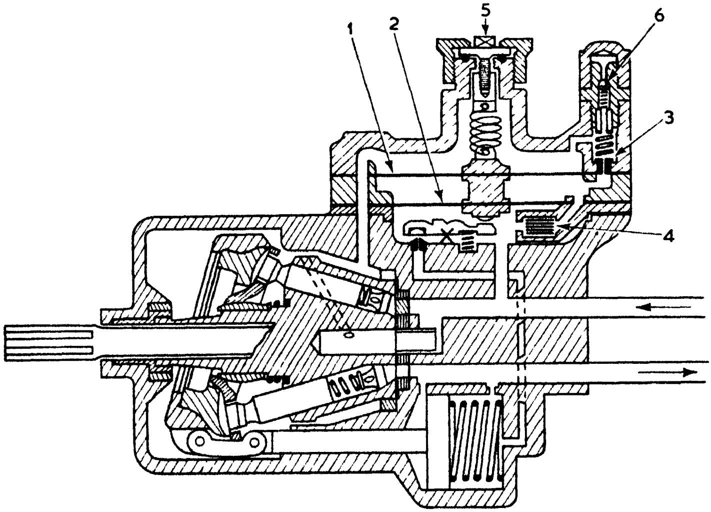
The loading of the springs on the control piston, opposing the movement of the cam plate to the zero position, is so arranged that the minimum pressure at the burners when the governor is in operation is more than adequate to maintain atomization of the fuel. The possible extinction of the flame due to governor action is thereby prevented. Should the rotational speed of the engine be lowered by ‘ throttling back ’ and reducing the fuel supply to the burners, the pressure acting on the diaphragm is reduced until contact between it and the rocker is broken and the plate valve is re-seated ; pressure above the control piston is restored and the cam plate is moved to increase the stroke. It is essential for correct functioning of the pump that no air is allowed to accumulate on either side of the diaphragm, and a ball-type bleed valve is provided to evacuate air from both upper and lower spaces. The ‘ C ’ size pump illustrated has plungers of 0·551 in diameter with a maximum stroke of 0·706 in and can be run up to 3,500 r.p.m. with gasoline or 4,000 r.p.m. with kerosine. The maximum rate of flow is 255 gal/hr per 1,000 r.p.m. At the maximum delivery pressure of 2,000 lb/in^2 the flow is 750 gal/hr and the necessary power input under those conditions, assuming an overall efficiency of 80 per cent, is 7·5 h.p./1,000 r.p.m. The dry weight is 14 lb.

Intended for the larger power units now under development is the ‘ D ’ size Mk. II pump delivering about 1,500 gal/hr. Especially it is useful for turbojets incorporating some means of thrust augmentation, such as afterburning or water injection, which in operation may almost double the rate of fuel consumption. It is of the same basic design as the 'C' size pump but the governor mechanism is arranged in axial alignment with the rotor, instead of at 90°, to yield the slim, cylindrical overall configuration particularly suitable for installation on axial-flow turbojets.

DENSITY-COMPENSATED GOVERNOR

While the simple hydraulic centrifugal governor was adequate for any of the fuels having a consistent or closely specified density, and could be re-set if a change of fuel became necessary or desirable, the introduction of wide-cut distillate fuel raised new problems. With this fuel a possible variation of 10 per cent in specific gravity may occur between one batch and the next and, at a given setting, the governor permitted too great a variation of the limited speed. Accordingly, there was developed a type of governor, shown diagrammatically in Fig. 7 : 3, which compensates automatically for density variation.

Compensation is provided by opposing to the centrifugal pressure component a differential pressure responsive to variation of centrifugal pressure and viscosity with change of specific gravity. A second diaphragm 2 of reduced area is mounted below the normal diaphragm 1 and the intervening space is in communication with the chamber above the diaphragm 1 by way of a square-law orifice 3. From the intervening space the fuel passes through a viscous flow orifice 4 to the governor spill chamber and thence to the pump inlet.



1. Upper diaphragm: 2. Lower diaphragm: 3. Square-law orifice: 4. Viscous flow orifice: 5. Setting adjustment: 6. Bleed valve.

Fig. 7 : 3—Density-compensated governor.

The proportional pressure between the two orifices, itself influenced by the fuel viscosity providing a commensurate pressure drop across orifice 4, will bias the diaphragm 1 against the centrifugal pressure on its upper surface. With a high-density fuel the centrifugal pressure component will be high and the proportional pressure will be high. Using a low-density fuel the converse will apply. Given a suitable setting the resultant force acting to open the pump servo valve will occur at approximately the desired rotational speed, irrespective of the actual fuel density.

BAROMETRIC CONTROL UNIT

As flying altitude is increased it is necessary to reduce automatically the delivery of fuel. The barometric pressure control unit varies the delivery pressure of the pump in accordance with change of altitude. As the pressure-control piston of the pump is connected to the cam plate, the stroke is varied and the delivery flow of fuel adjusted to specific requirements.

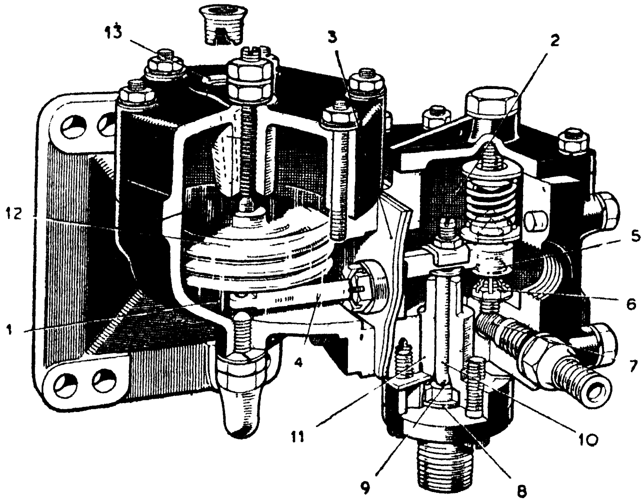
Constructionally, the unit comprises a body divided into a capsule chamber and a valve chamber by a diaphragm which isolates the two chambers and serves as a pivot plate to which is secured a rocker lever extending into both. The barometric capsule is mounted with its lower pivot seated in one end of the rocker lever and its upper pivot located in an adjustable screw mounted in the chamber cover.

The other end of the rocker lever is loaded by a spring and carries a half-ball seating on an orifice supplied with fuel at servo pressure from above the pressure-control piston of the pump. Control of the rocker lever is by fuel at delivery pressure from below the pressure-control piston. Pressure is applied to a rubber diaphragm and transmitted by a small piston and an operating rod, the upper end of which abuts the end of an adjuster screw in the rocker arm. These components are mounted in the base of the chamber in an eccentric sleeve to enable the end of the rod to be moved across the face of the adjuster screw to alter the operative distance from the fulcrum point of the lever. The valve chamber is in open communication with the inlet side of the pump.

Should the force applied by the pump delivery pressure exceed the spring loading the rocker lever will pivot, lift the half-ball from the orifice and allow servo pressure fuel to spill into the valve chamber. This unbalances the pressure-control piston, which moves to reduce the stroke of the pump. Consequently the pump output decreases, and the delivery pressure is lowered until it just suffices to balance the spring load. Balance is maintained by a small spill past the half-ball. A fall in pump delivery pressure allows the spring to seat the half-ball, closing the orifice and, in turn, increasing the pump stroke.

The barometric capsule exerts a force which assists the pump delivery pressure to oppose the spring load. At ground level the force is determined by adjustment of the upper pivot screw which compresses the capsule. As altitude is increased and barometric pressure falls, the capsule tends to lengthen, and the pressure exerted on the rocker lever becomes greater. The spring load remains constant, and consequently the pump delivery pressure reduces until a balance is reached.

As the barometric control unit is mounted on the wheel-case on the front of the compressor casing it is inside the nacelle and consequently responsive to the ram effect at speed as well as altitude and local atmospheric conditions.



1. Capsule chamber: 2. Valve chamber: 3. Pivot plate: 4. Rocker: 5. Half-ball. 6. Orifice: 7. Filter. 8. Diaphragm: 9. Piston. 10. Operating rod· 11. Eccentric adjusting sleeve. 12. Barometric capsule: 13. Vent to atmosphere.

Fig. 7 : 4 —Barometric Pressure Control Unit.

THROTTLE VALVE

From the pumps the fuel is delivered to the pilot's throttle valve, a manually-operated variable orifice, and thence to the high-pressure cock, also manually operated by the pilot to cut the supply to the burners and so shut down the unit. On the throttle valve is an adjustable by-pass through which fuel flows to maintain the engine running at idling speed when only the throttle valve is closed.

SIMPLE FLOW CONTROL UNIT

A subsequent development in control systems was the introduction of a unit known as the Simple Flow Control. Designed to economize in space and eliminate sources of external leakage, this component groups

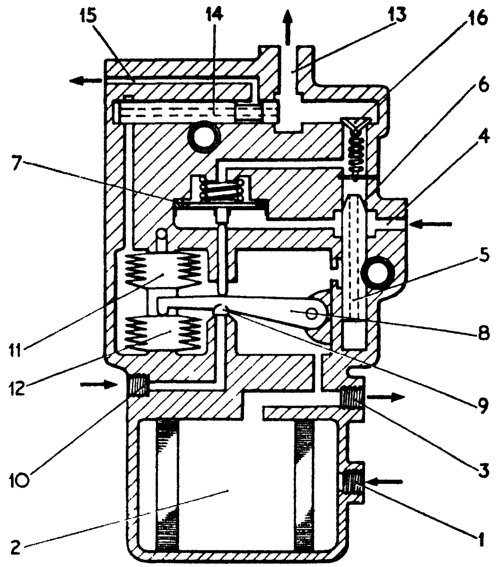


Fig. 7 : 5—Simple Flow Control Unit.

1. Fuel inlet.
2. Filter.
3. Fuel to pump.
4. Fuel from pump.
5. Throttle plunger.
6. Metering orifice.
7. Piston.
8. Rocker lever.
9. Servo valve.
10. Pump servo connection.
11. Nacelle pressure capsule.
12. Evacuated capsule.
13. Fuel to burners.
14. Shut-off cock.
15. Dump to atmosphere.
16. Priming valve.

together a throttle valve, shut-off cock, low pressure filter and an altitude-responsive flow control device. An important advantage in the use of this unit is the reduction in pump pressure at part throttle openings. The pump is thus called upon to perform less arduous duty than when used in conjunction with a barometric control unit and consequently its effective life is lengthened. In its standard form the unit can handle fuel flows up to about 500 gal/hr and can reasonably match turbine requirements at full or predetermined throttle openings up to altitudes of approximately 30,000 ft.

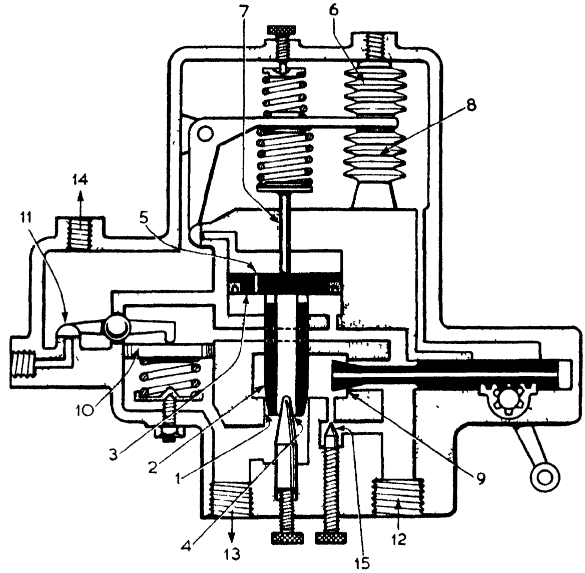
The functioning of the flow control can be followed by reference to Fig. 7 : 5. A throttle valve plunger with a suitably contoured end controls a variable metering orifice in the fuel delivery line. Movement of this plunger is effected by a manual control in the pilot's cockpit rotating a pinion engaging rack teeth cut in the side of the plunger. As the area of the metering orifice is changed the pressure drop through the orifice is also varied. This pressure difference across the throttle is applied to a spring-loaded piston, movement of which is transmitted through a push-rod to a rocker lever carrying a valve controlling a passage communicating with the pump servo-system. The end of the rocker is engaged by a pair of capsules, the upper open to nacelle pressure and the lower evacuated, thus rendering the system responsive to atmospheric pressure. If a given pressure difference is exceeded at the metering orifice, and consequently on opposite sides of the piston, the piston will be moved against its loading spring and the rocker lever will lift to spill momentarily pump servo-pressure. As a result the pump control piston will move to decrease the stroke of the pump pistons and reduce the delivery of fuel.

At increased throttle settings the pressure drop across the metering orifice will be reduced and the pressures on opposite sides of the piston will be more nearly similar. The piston will then move under the influence of its spring and in turn more pressure will be exerted on the rocker lever, holding the servo-valve closed and causing pump delivery to be increased until the pressure drop across the metering orifice is restored. An increase in nacelle pressure due to ram effect or denser atmospheric conditions will cause the upper capsule to expand and the loading of the servo-valve to be increased, leading to an enhanced delivery from the pump. Conversely, a decrease in nacelle pressure due to lower aircraft speed, increased altitude or lower barometric conditions will have an opposite effect. When it is desired to stop the engine, the shut-off cock, of the by-pass type, is moved and fuel in the burners, manifolds and pipe is dumped to atmosphere. A priming valve located downstream from the throttle valve has two functions. It is set to open at about 70 lb/in^2 which ensures there is always sufficient pressure for satisfactory operation of the pump servo-system and, additionally, it maintains the fuel system primed up to the shut-off cock when the engine is stationary.

FULL-RANGE FLOW CONTROL UNIT

The operating requirements of power units for the later types of aircraft tend to exceed the capacity of the Simple Flow Control and, to meet the enhanced demands, a Full-range Flow Control system has been developed. This can function effectively to altitudes of the order of 60,000 ft and can handle fuel flows of 1,000 gal/hr. The pressure drop across the unit—from pump delivery pressure to burner pressure—is regulated by the nacelle pressure and the setting of the throttle valve under the manual control of the pilot. Nacelle pressure is taken at the intake to the compressor of the power unit and varies with atmospheric conditions, altitude and speed.

The area of the main metering orifice 1 (Fig. 7 : 6) is varied by the contoured end of a plunger 2 forming the stem of servo piston 3. In conjunction with an adjustable contoured plug 4, the bore of the plunger provides a metering orifice for idling conditions. Fuel is supplied to the underside of the servo piston and passes by way of a restricting orifice 5 to the upper side from which it can be released to the main casing and thence returned to the pump inlet. Release is controlled by a half-ball valve carried by a bell-crank lever under the opposing influences of the nacelle pressure capsule 6 and a spring-loaded push rod 7 from the servo piston. The other capsule 8, is evacuated. Under normal operational conditions the bell-crank lever is balanced by the opposing forces and the half-ball valve is just floating.



1. Metering orifice: 2. Plunger: 3. Servo piston: 4. Idling orifice: 5. Restricting orifice: 6. Nacelle pressure capsule: 7. Servo push rod: 8. Evacuated capsule: 9. Throttle orifice: 10. Pressure drop piston: 11. Pump servo valve: 12. Inlet from pump: 13. Outlet to burners: 14. Return to pump inlet: 15. Trimming orifice.

Fig. 7 : 6—Full-range flow control unit.

The position of the servo piston, varied by change of nacelle pressure, determines the areas of the metering orifices and these, together with the selected area of the throttle orifice 9, affect the pressure drop across the unit. This difference in pressure is exerted on opposite sides of a piston 10 spring-loaded to maintain the pump servo valve 11 closed. When the pressure drop is sufficient to overcome the loading spring the piston moves to lift the rocker and open valve 11 to spill pump servo pressure fuel to the casing and back to the pump inlet. As a result the pump control operates to reduce the stroke and the delivery from the pump.

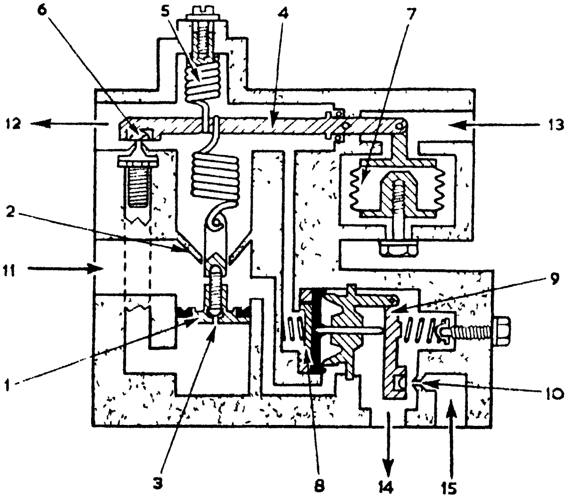
Any tendency for the pump to deliver an increase of fuel will increase the pressure drop and piston 10 will operate to reduce the delivery. With increased throttle opening the pressure drop will be reduced and piston 10 will hold valve 11 tightly closed so that pump delivery is increased until the pressure drop is built up again and equilibrium is re-established.

AIR : FUEL RATIO CONTROL UNIT

Serious risk of mechanical failure may be incurred if gas temperatures in the combustion chambers or at the turbine entry are permitted to rise above designed maxima. Such excessive temperatures may occur on sudden acceleration if additional fuel is fed more quickly than the air mass flow from the compressor is increased. Conversely, over-fuelling may arise if the power unit is operated in conditions under which surging or stalling of the compressor can occur, involving a reduction of the air mass flow. The purpose of this unit is to prevent the occurrence of excessive temperatures by exercising an overriding control on the fuel flow system. This is effected by regulating the delivery of fuel in accordance with the compressor delivery pressure, since the air mass flow varies directly with the absolute compressor pressure.

Referring to the schematic arrangement in Fig. 7 : 7, a servo piston 1 has a stem furnished with tapered slots to form a flow plunger in conjunction with an orifice plate 2. Transverse drillings in the stem communicate with an axial bore housing a fine filter and terminating in a calibrated restricting orifice 3 opening to the chamber above the piston head. The stem is coupled to a rocker lever 4 by a helical spring and a second spring assembly 5 affords balance and a means of adjusting the loading of the lever. One end of this lever carries a half-ball to seal a filter-screened orifice 6 in communication with the chamber above the servo piston while the other end, beyond a sealed fulcrum point, is hinged to an evacuated capsule 7 in a chamber receiving compressor delivery pressure.

In a third chamber a piston 8 with a skirted seal is under the influence

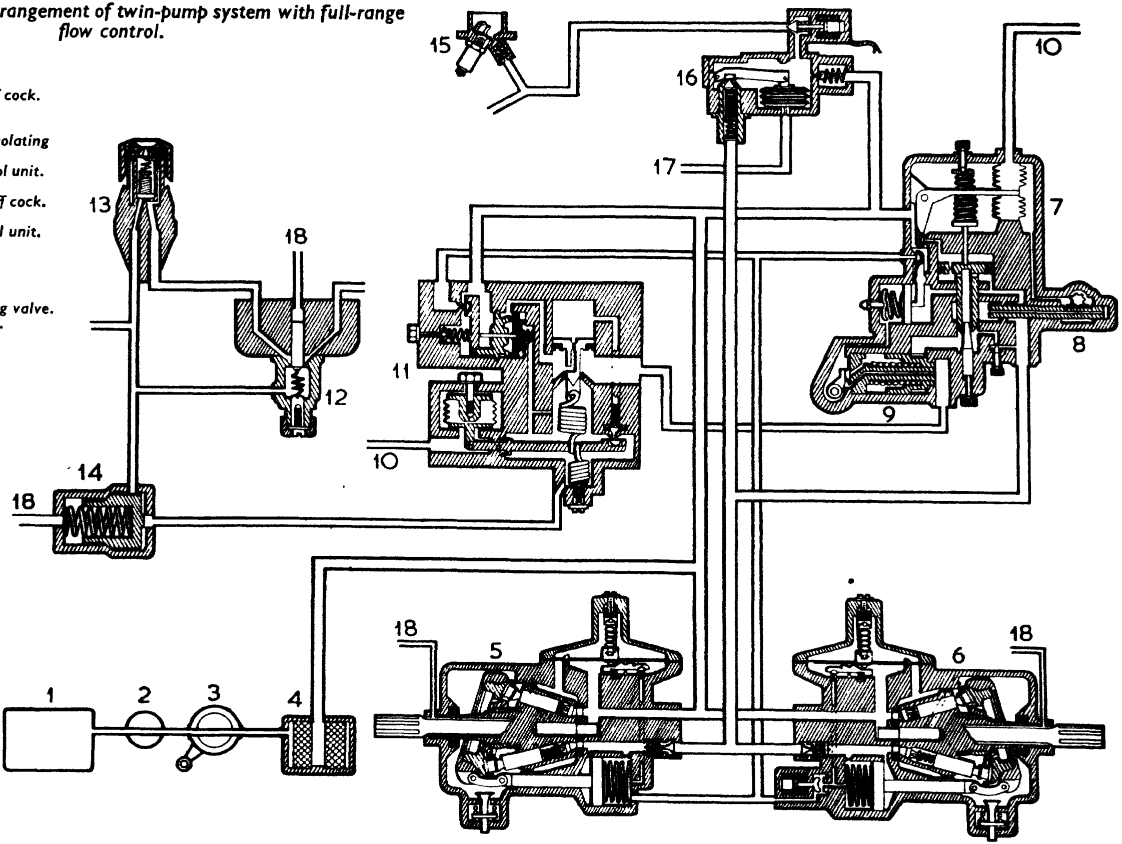


1. Servo piston: 2. Orifice plate: 3. Restricting orifice: 4. Rocker lever: 5. Load-adjusting spring: 6. Screened orifice: 7. Evacuated capsule: 8. Pressure drop piston: 9. Hinged lever: 10. Pump servo orifice: 11. Fuel inlet: 12. Fuel to burners: 13. Compressor delivery pressure: 14. Pump servo inlet. 15. Return to pump inlet.

Fig. 7 : 7—Air : fuel ratio control unit.

Fig. 7 : 8—*Typical arrangement of twin-pump system with full-range flow control.*

1. Fuel tank.
2. Booster pump.
3. Low-pressure shut-off cock.
4. Filter.
5. Supply pump.
6. Supply pump with isolating valve.
7. Full-range flow control unit.
8. Throttle control.
9. High-pressure shut-off cock.
10. Nacelle pressure.
11. Air : fuel ratio control unit.
12. Flow divider.
13. Duplex burner.
14. Dump valve.
15. Torch igniter.
16. Torch igniter reducing valve.
17. Atmospheric pressure.
18. Drain to atmosphere.



of the main inlet (delivery) pressure on the underside and, supplementing the loading spring, on the other side, the burner fuel pressure. On the higher pressure side the piston abuts a push rod resting on a hinged lever 9 carrying a half-ball controlling an orifice 10 in communication with the fuel pump servo control system.

Compressor delivery pressure, proportional to the air mass flow, translates into mechanical force due to its compression of the capsule. This force acting on the rocker lever 4 is opposed by the tension spring, so the spring force becomes proportional to compressor pressure. Since the spring has a linear rate of extension and the slotted stem of the servo piston directly varies the effective area through the orifice plate 2, the maximum fuel flow is made proportional to the air mass flow. The position of the servo piston is stabilized when the half-ball over orifice 6 just floats to release fuel from the servo piston chamber at the same rate as it is made up through the restricting orifice 3.

Additional to its main function, the air : fuel ratio control also serves to ensure there is no risk of rich-mixture extinction occurring at high altitudes.

BURNERS

There are two main types of pressure-atomizing burners, designated Simplex and Duplex The first is of the fixed-orifice type with a simple swirl atomizer in which the pressure is proportional to the square of the fuel flow. Fuel is passed through a tubular gauze strainer (supported by a spring so that if completely clogged it will lift and be by-passed) to the interior of the burner body, in the end of which is mounted the atomizer. From the annular space between burner bore and atomizer the fuel enters the conical atomizer chamber by three tangentially arranged holes, producing a rapid swirl. At the apex of the cone is a precisely calibrated orifice through which the fuel is projected in the form of a thin film which breaks up into a finely-atomized, readily ignitable spray. The atomizer is secured to the burner by a flanged sleeve drawn up by the air shroud. The function of the shroud is to pass a controlled flow of air

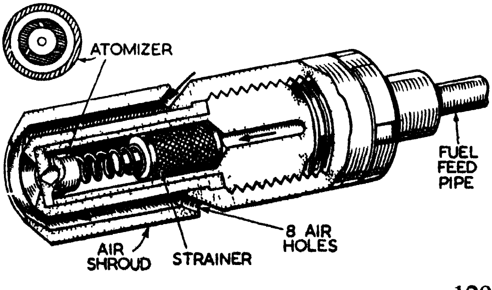


Fig. 7 : 9—Simplex burner of fixed-orifice type.

to the face of the atomizer to prevent the formation and build-up of carbon.

Atomizers are graded in size according to flow number, the flow number FN being equal to $\frac{Q}{\sqrt{Pd}}$ where Q is the quantity of fuel, gal/hr and Pd the pressure drop, lb/in². For example if $Q = 20$ gal/hr and $Pd = 400$ lb/in², then flow number $= \frac{20}{\sqrt{400}} = 1{\cdot}0$. With, say, ten burners each having a flow number of 1·0, connected in parallel, the overall flow number will be 10·0. Three basic sizes of burner are produced with the following approximate range of flow numbers: Large—1·35–3·5, Medium—0·45–1·35, and Small—0·3–0·45.

With atomizers of this type the effective range of fuel is relatively limited if an excessively high pressure at one end or an unsatisfactorily low pressure at the other end is to be avoided. This disadvantage is overcome by the use of the Duplex burner which can deal with the wide range of fuel flows demanded if operation to very high altitudes is required.

DUPLEX BURNERS

A pressurizing valve (Fig. 7 : 1) is inserted between the high-pressure cock and the ring manifolds feeding the burners. This is a spring-loaded, conical-seating plunger which at low pressure and volume is closed to ensure that all fuel is delivered by way of the small-diameter pilot manifold. As pressure builds up the valve is lifted against its spring to admit fuel to the larger diameter main manifold. Incorporated with this unit is the spring-loaded dump valve. In operation, even at idling speeds, the fuel pressure closes this valve, but when the high-pressure cock is shut off and pressure falls, it opens automatically to drain the manifolds.

The provision of Duplex fuel burners is to ensure, without resort to abnormally high pressures, a completely atomized fuel spray at the low rates of flow occurring when cranking for a start and also under high-altitude conditions. Separate connections from the pilot and main manifolds are taken to each burner. Under reduced flow conditions all the fuel is delivered to the pilot atomizer. It passes through helical grooves cut in the face of a conical plug to reach the swirl chamber, and is projected as a finely atomized conical spray.

At the substantially higher rates of flow during normal operating conditions, the pressurizing valve is lifted and fuel is delivered to both the pilot and the main manifolds. The pilot flow continues as before, whilst the fuel from the main manifold passes to an annular space surrounding the housing of the primary swirl cone. Through a series of holes in the ring element in which the primary swirl cone is seated the fuel reaches the swirl plate having a number of tangential slots communi-

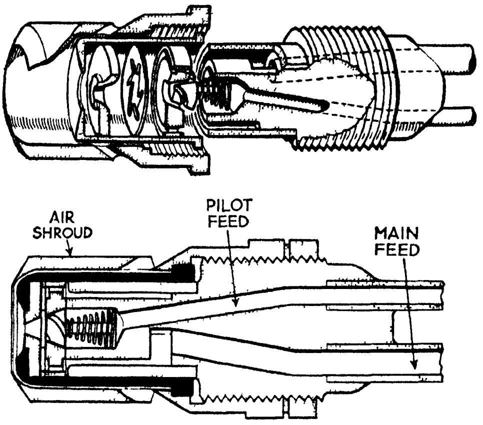


Fig. 7 : 10—In the Duplex I burner primary and main flows are delivered to a common swirl chamber.

cating with the central swirl chamber. Here the primary and main flows mix and pass out of the final orifice. The functions of a particular burner may be readily adjusted by substituting a swirl plate of different thickness which, of course, alters the volume of the swirl chamber.

This type of burner, Duplex 1, requiring two manifolds and a pressurizing valve, is shown in the system layout in Fig. 7 : 1. A later type, termed Duplex 2, needs only a common feed for both primary and main fuel supplies as a relief-type pressurizing valve is incorporated in the assembly. It will be noted that the swirl cone is absent in this design and that primary and main flows have each a swirl plate communicating with a common swirl chamber. The valve is usually set to open at 90–100 lb/in^2 so that at idling speeds at ground level, and at reasonably high r.p.m. at altitude, the flow is through the primary system only and good atomization is ensured.

The latest-pattern burner, Duplex 3, reverts to individual feed lines for primary and main flows and a single pressurizing valve unit. Each system has its individual swirl plate and swirl chamber, the main chamber being annular and enshrouding the primary chamber. Thus better atomization at low rates of primary flow is ensured. The primary flow of fuel is through a filter, two holes in a distributor block, the primary swirl plate, and the swirl chamber in the rear orifice plate. This chamber projects through the centre of the main swirl plate and into the main swirl chamber.

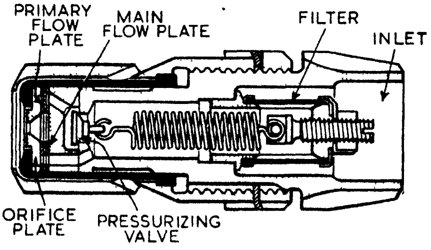
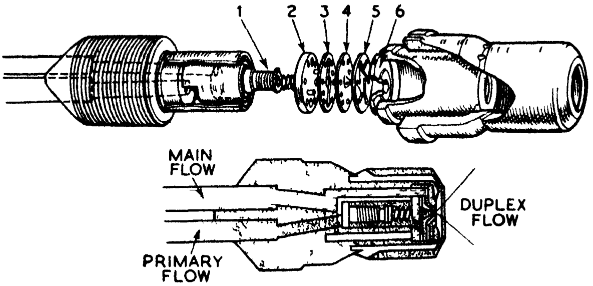


Fig. 7 : 11—Arrangement of Duplex 2 burner with incorporated pressurizing valve.



1 Filter 2 Distributor block 3 Primary swirl plate 4 Rear orifice plate 5 Main swirl plate 6 Front orifice plate

Fig 7 · 12—Duplex 3 burner has separate primary and main swirl systems

When the pressurizing valve opens, the main fuel flow is through a series of six holes in the distributor block and primary swirl and orifice plates to the main swirl plate and the main swirl chamber in the front orifice plate

MINIMUM BURNER PRESSURE UNIT

This unit is employed to ensure that fuel flow to the burners never falls to a value such that flame extinction can occur due to poor atomization. A capsule acts through a rocker lever and a half-ball orifice to control the fuel inlet and thereby the pressure in the reducing chamber. Fuel from the pump is admitted by a banjo connection to the main inlet fitted with a filter and terminating in the half-ball orifice A relief valve outlet is connected to the low pressure side of the fuel system and the minimum burner pressure valve connects the reducing chamber with the burner manifolds. The unit is thus in parallel with the throttle valve

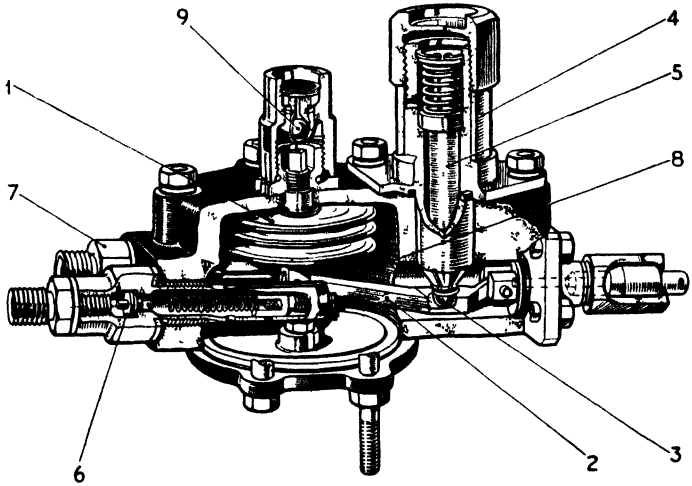
Under certain conditions the pressure downstream of the throttle valve may fall to 2 or 3 lb/in^2 and at such values satisfactory atomization is not possible. Since fuel is delivered to the unit at pump pressure however, the inlet pressure will not fall below about 100 lb/in^2. Fuel pressure in the reducing chamber compresses the capsule and causes the rocker lever to rise and close the half-ball orifice when a pressure of about 35 lb/in^2 is reached. If the capsule is open to nacelle pressure this figure will be a rate varying with altitude through about 10 lb/in^2 in 40,000 ft. A gas-filled or evacuated capsule will maintain the chamber fuel pressure constant, apart from slight deviations due to temperature changes.

Should the pressure downstream of the throttle valve fall below 10 lb/in^2, the minimum burner pressure valve, which is set to lift with a pressure drop of 25 lb/in^2, will open to admit a supply of fuel to the burner manifold, by-passing the throttle valve. A ball-valve is provided to prevent leakage of fuel from the reducing chamber in the event of damage occurring to the capsule. To protect the capsule against excessive pressure the relief valve opens at reducing chamber pressures in excess of 45 lb/in² to spill fuel to the low pressure side of the system.

With a full-range flow control system a minimum burner pressure unit is not needed but in these systems the unit is employed as a reducing valve for the torch igniters as in Fig. 7 : 8. A shut-off cock is incorporated which is actuated by a solenoid energized to open the outlet from the unit for the starting routine. The pressure is commonly set at about 35 lb/in²

DUAL PUMP SYSTEM

One pump may be insufficient to meet the requirements of the larger power units and in other instances it may be more convenient to fit two pumps rather than a single large one. Duplication tends to increase safety in operation. In dual pump systems an additional safety factor can be provided for conditions at take-off and landing. By the use of a solenoid-operated valve the control systems of the two pumps can be isolated in the event of failure in one of the pumps, the altitude control unit or the servo connecting pipes, and full delivery obtained from one pump. The usual arrangement is for each pump to be capable of delivering sufficient fuel for take-off and for a wide range of operational requirements. Each pump has the standard hydraulic control mechanism to limit delivery and consequently turbine speed to a safe maximum and each has a non-return delivery valve so that fuel at high pressure cannot be passed back to the



1. Capsule.
2. Rocker lever.
3. Half-ball orifice
4. Fuel from pump.
5. Filter.
6. Relief valve.
7. Minimum burner pressure valve.
8. Reducing chamber.
9. Ball valve.

Fig. 7 : 13—Minimum Burner Pressure Unit.

failed pump. The governing devices are set to slightly different values to prevent the two units striving for control.

One pump only is fitted with an isolating valve. Should this pump fail the control pistons of both pumps, interconnected by a common pipe, will move towards the zero delivery position until the isolating valve, under pilot control, is closed by energizing the solenoid. This severs the interconnection, the faulty pump operates on zero stroke, the non-return valve prevents the circulation of fuel, and the sound pump maintains full delivery and also servo pressure to the altitude control unit.

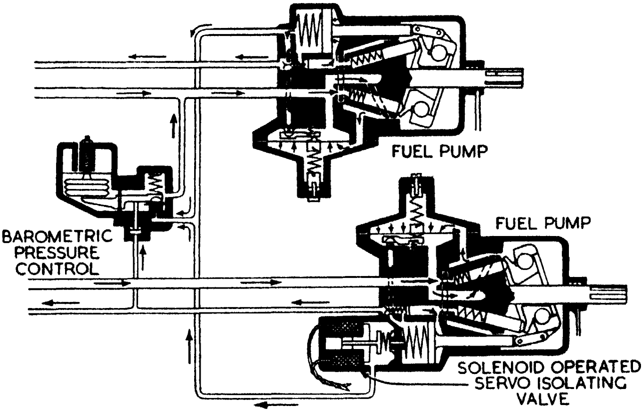

FUEL PUMP
FUEL PUMP
BAROMETRIC PRESSURE CONTROL
SOLENOID OPERATED SERVO ISOLATING VALVE

Fig. 7 : 14—Dual pump system with isolating valve.

In the event of failure in the pump not fitted with the isolating valve, in the altitude control unit or in the interconnecting pipe, initial effect will be the same but the closing of the isolating valve will result in the sound pump operating independently of the altitude control unit.

The most serious occasions when a pump failure might occur are during take-off or when attempting another circuit after a baulked landing. Accordingly, the isolating valve is actuated prior to take-off or landing so that one pump is used, operating at maximum governed speed, to avoid the danger of time lag in pick-up speed.

TORCH IGNITERS

A minor problem of ignition is the location of the sparking plug in relation to the fuel spray. To secure an effective position the end of the plug was usually more or less exposed to the heat of combustion and liable to suffer accordingly. A solution was found in the provision of what is termed the torch igniter. This is a small self-contained unit embodying a special low-pressure atomizer and a sparking plug. Fuel is supplied at booster pump pressure and controlled by a solenoid-operated valve on the igniter (Fig 7 : 1) or is fed from the reducing valve and controlled by the solenoid-operated valve on that unit (Fig. 7 : 8). Normally this valve is closed, but at appropriate timing in the starting cycle the solenoid is energized and fuel flows through the atomizer and is ignited by the sparking plug. A flaming jet of fuel, as from a blow torch, is then projected into the fuel spray from the burner in the combustion chamber. The ends of both atomizer and sparking plug are shrouded, and in any case are only exposed to the secondary air in the annular space between the flame tube and the outer casing of the combustion chamber. One igniter is sufficient for a complete engine unit, but two are fitted to cover all eventualities. From the two combustion chambers fitted with torch igniters combustion is propagated to other chambers through the usual communicating pipes. The igniters, of course, only function for a brief period when starting, and during normal operation are completely cut off.

For military machines it is necessary to make arrangements for exceptional conditions during aerobatics. The fuel tank is fitted with an inverted-flight valve to ensure an uninterrupted supply of fuel under conditions of ‘ negative g.’ Fig. 7 : 16 shows the flow under two different conditions. At about mid-height, the tank has a horizontal partition furnished with flap valves closing on the under side. The outlet tube has a lower orifice on the tank base and an upper orifice adjacent to the partition. Over the tube slides a sleeve actuated by a weight and linkage to control the lower orifice. In normal operation the sleeve is raised

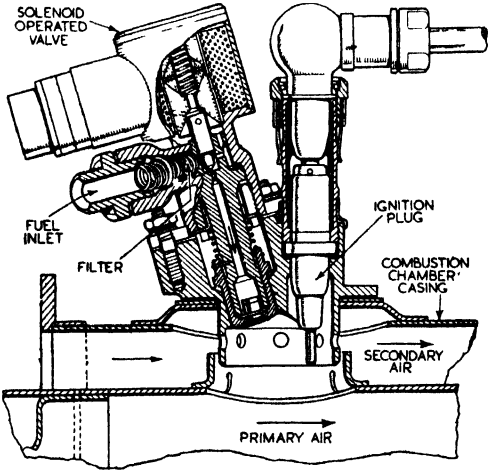


Fig. 7 : 15—Torch igniters are fitted to two combustion chambers only.

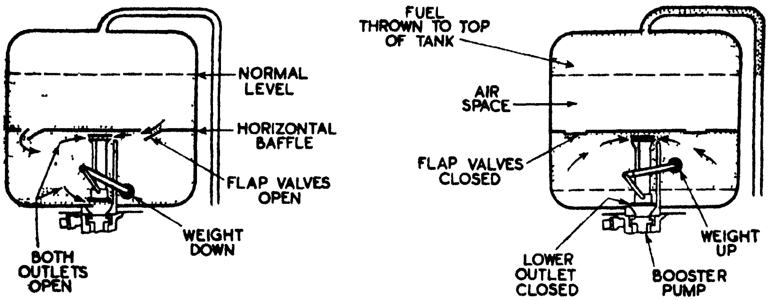


Fig. 7 : 16—The inverted-flight valve in the fuel tank. Left, normal conditions. Right, ' negative g ' conditions.

by the weight and fuel flows through the flap valves in the partition and through both upper and lower outlets to the Self-Priming booster pump. Under conditions of ' negative g,' when the fuel tends to surge to the top of the tank, the flap valves close and some portion of the fuel is trapped in the lower compartment. The weight rises, closes the lower orifice, which is now exposed, and fuel continues to flow from the upper orifice below the partition.

STARTING CYCLE

Starting is effected by a 24-volt electric motor and all necessary operations are effected by a timed automatic cycle. When the master switch is closed, current is supplied to the booster fuel pump and a warning light glows until the requisite pressure is built up. With the throttle closed, depression of the starter button brings the clockwork time switch into operation. In timed sequence, current is supplied to the ignition coils, torch igniters and the starter motor. The motor takes up the drive and cranks the engine at a low speed for about 5 sec, after which a relay closes to allow full excitation of the motor. The time switch completes its cycle in 30 sec and cuts off all current. If, however, the engine starts and reaches a predetermined speed prior to this, starter, ignition coils and flame igniters are automatically cut out.

DE HAVILLAND SYSTEM

Early Goblin units had a fuel system incorporating a constant-stroke pump and a barostat, or automatic altitude control, to spill excess fuel back to the tank. From the tank the fuel is fed through a filter to a Dowty seven-plunger constant-stroke pump which delivers to a control unit embodying a manually-controlled, tapered-needle type throttle valve. Here the fuel is divided, with the main supply passing to a rotary distribution and shut-off cock and a second supply through a high-pressure filter to the barostat. From the distribution cock the fuel passes through a hydraulic valve under the control of an engine-driven, over-speed governor to the starting valve and hence to the ring manifold and by individual flexible pipes to the burners.

Control is by variation of the fuel pressure, and consequently the flow at the burners. This is effected by manual operation of the throttle lever which adjusts the position of the tapered needle in the control box. Closing movement of the needle is limited by an idling stop, so the unit cannot be inadvertently shut down by the throttle lever.

BAROSTAT

The barostat, or altitude control unit, is essentially a relief valve, the setting of which is related to barometric pressure and hence to altitude. The unit incorporates a relief valve loaded by a stack of co-axial springs, a piston serving as a variable datum abutment for the springs, a barometric capsule and a pilot valve controlling the admission of fuel to the upper face of the piston. A proportioning valve supplies high-pressure fuel to the pilot valve system and a ball-type relief valve limits the pressure from the proportioning valve to 50 lb/in^2.

Fuel from the pump is admitted to the underside of the relief valve and, as pressure increases, the constraint of the springs is overcome and the valve opens to by-pass fuel by way of a return pipe to the fuel tank. The loading of the springs is determined by the position of the piston

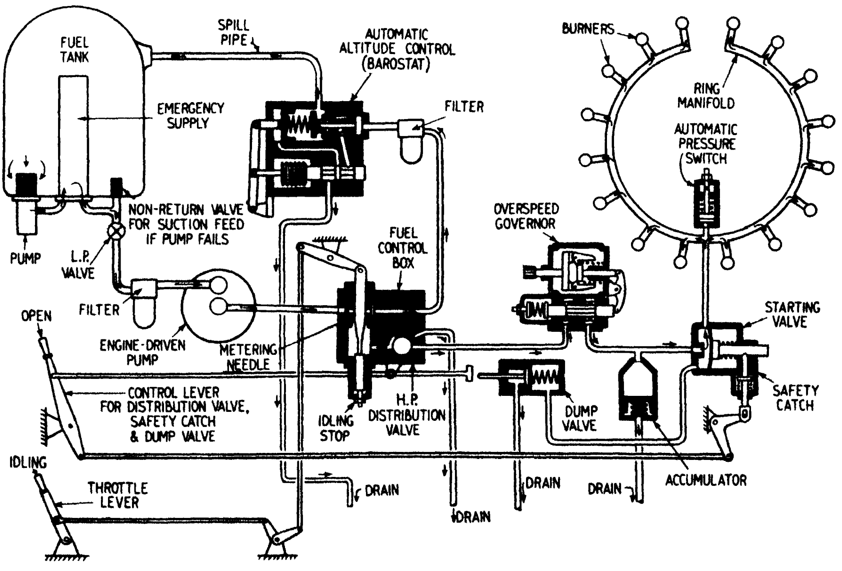
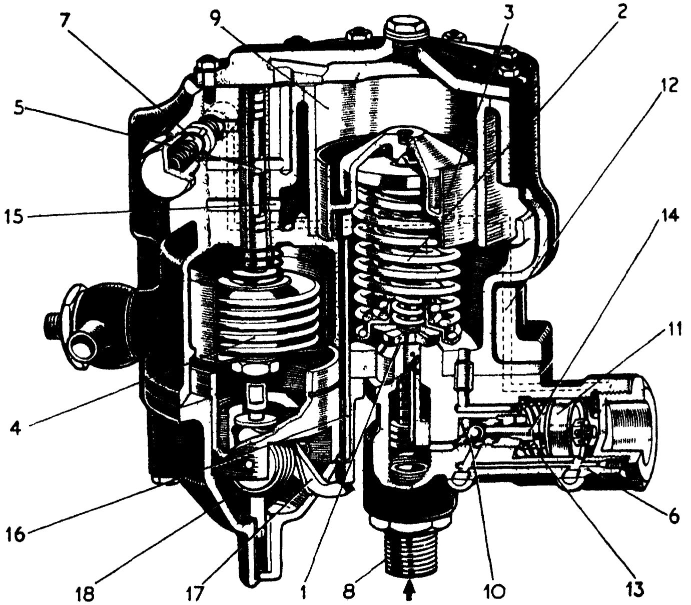


Fig. 7 : 17—Early Goblin fuel and control system.

which is controlled by admission or escape of fuel to or from the cylinder. Fuel from the inlet reaches the pilot valve by means of a port, thence across the top of the proportioning valve piston and through a connecting passage. Pressure builds up above the proportioning valve piston until the loading spring and the fuel pressure on the valve plunger is overcome and the port is closed. Thus a state of balance is established between proportioning and pilot valves that is proportional to the pressure at the inlet.

An increase in altitude and corresponding decrease in barometric pressure allows the capsule to expand and move the pilot valve upwards to connect the cylinder with the spill port. Relieved of pressure on its upper face, the piston moves up to reduce the spring loading on the relief valve. This movement is accompanied by a corresponding downward movement of the pilot valve by means of the mechanical linkage; rod, lever and spring. The spill connection is thus cut off and the piston stabilized in its new position.



1. Relief valve: 2. Springs: 3. Piston: 4. Barometric capsule: 5. Pilot valve: 6. Proportioning valve: 7. Ball relief valve: 8. Fuel from pump: 9. Cylinder: 10. Port: 11. Proportioning valve piston: 12. Pilot connecting passage: 13. Proportioning valve spring: 14. Proportioning valve plunger: 15. Spill port: 16. Rod: 17. Lever: 18. Spring.

Fig. 7 : 18—Barostat.

With a decrease in altitude the capsule is compressed and moves the pilot valve downwards to connect the pilot passage with the cylinder.

This permits a flow of fuel into the cylinder, forcing down the piston, increasing the loading of the relief valve and reducing the spill back to the tank.

STARTING ROUTINE

The throttle is closed to the idling position, and by means of a second lever the shut-off cock is opened. Operation of a push-button closes the circuits for the starter motor, the booster coils for the ignition plugs, and the time switch. Thereafter the sequence is automatically controlled. The motor is operated at low speed through two resistances for 5 sec to ensure easy engagement. Then one of the resistances is cut out, the motor speed increases and the main jet unit is brought to a speed sufficient to raise fuel pressure against the closed starting valve. A spring-loaded accumulator in the fuel line between the governor and the starting valve is charged with fuel, and when a predetermined pressure is reached the starting valve lifts and is then retained in the open position by a spring-controlled safety catch. Fuel then passes to the manifold and the burners and ignition is effected.

Simultaneously, a pressure-actuated switch in connection with the fuel manifold cuts out the second resistance, thereby releasing more electrical energy for the starter motor which continues to drive and assists the acceleration of the main jet unit. Thirty seconds after operation of the push button the time switch cuts off the motor and the ignition system.

To stop the unit, it is brought down to idling speed with the throttle lever and then shut off by the second control lever. Operation of the shut-off lever cock releases the safety catch on the starting valve and opens the dump valve, which permits the fuel in the manifold to drain to atmosphere.

In the event of a false start it might be possible for fuel to accumulate in the heads of the three lower combustion chambers. For safety, therefore, the lowest combustion chamber is fitted with a valve which opens automatically when pressure in the chambers falls below 3 lb/in^2 and, through the interconnections, drains the affected three lower chambers.

DOWTY SPILL-BURNER SYSTEM

Use of the Dowty pump on the de Havilland Goblin turbojet led to co-operation between the two companies and to studies for a complete fuel system based on the spill-burner. In the strongly held view of Dr Moult, Chief Engineer of de Havilland, that type was the most likely to repay intensive development and be able to meet the requirements of the more advanced turbine power units of both the immediate and the more distant future.

As evolved, the system employs two positive displacement pumps

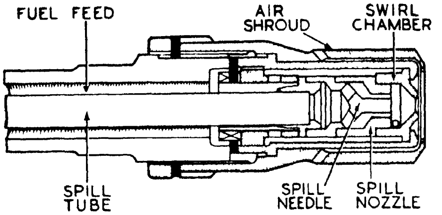


Fig. 7 : 19—Dowty spill burner

driven at about one-third turbine speed. One circulates fuel through the burner and the spill circuit and the other, a variable output unit, delivers a metered quantity of fuel into the spill circuit for discharge from the burner nozzle. The spill burner resembles the Simplex burner in that the fuel is fed through tangential ports into a conical swirl chamber having an axial discharge orifice but differs in the provision of an annular spill orifice in the base wall.

In Fig. 7 : 20 the functioning of the atomizing element is shown diagrammatically. Fuel enters the swirl chamber by two tangential ports and forms a free vortex, the pressure in which falls as it approaches the axis, while the angular velocity increases as it approaches the apex of the cone. By carefully proportioning the diameter of the discharge orifice and the inner diameter and the area of the spill annulus, it is possible to establish a condition of equilibrium in which the air core of the vortex is of greater diameter than the discharge orifice and all the fuel input is spilled back

By throttling back or otherwise impeding the spill flow, pressures will rise in the swirl chamber, the air core will be reduced in diameter and a thin film of fuel will be delivered from the discharge orifice in the form of a hollow cone breaking into a finely atomized spray.

At any given pump speed the burner can spill the full delivery of the circulating pump which, at that speed, can only take in the same volume as is delivered. It follows that the volume of fuel delivered by the supply pump to the spill line will necessarily be discharged from the burner. The effect is that of a variable throttle in the spill line but without the concomitant restriction.

A number of advantages are claimed for a spill system as outlined :

1. At any given pump speed atomization and combustion efficiency are substantially maintained irrespective of altitude. Actually, as the delivery at the burner is reduced at altitude, the discharge at the burner nozzle occurs at a higher pressure giving improved atomization.
2. As no small orifices are necessary in the nozzle, filtering arrangements may be less meticulous and, virtually, the system is not ‘dirt conscious.’

3. Construction is simplified and detail components can be produced by normal standards to permit interchangeability.
4. The circulation of fuel through the burner has a beneficial cooling effect when operating at high altitudes and consequent low discharge rates. This is of particular value where burners are arranged for upstream injection.

Both pumps are of the 14-cylinder, double-bank, radial type. The cylinder spool rotates on an axially-positioned fixed distribution valve and the plungers, each with a slipper pivotally attached, are flung outwardly to take a bearing on an eccentric track ring. While the circulating pump is built into the valve block, the supply pump (Figs. 7 : 21 and 7 : 22) incorporates the universal fuel governor. This unit follows the lines of the usual hydraulic overspeed governor but is elaborated to give automatic compensation for fuels of any density from aviation gasoline to light diesel oil.

An orthodox governor can be set to handle a fuel of given density and any variation in density arising from temperature changes will only affect the governed speed to a negligible extent. Should in emergency a change be made to a fuel of different density, however, there exists the possibility of a dangerous overspeeding or, conversely, a speed limitation that could hamper a take-off. Similar effects may well occur when the fuel is 'wide-cut distillate,' the specific gravity of which may vary

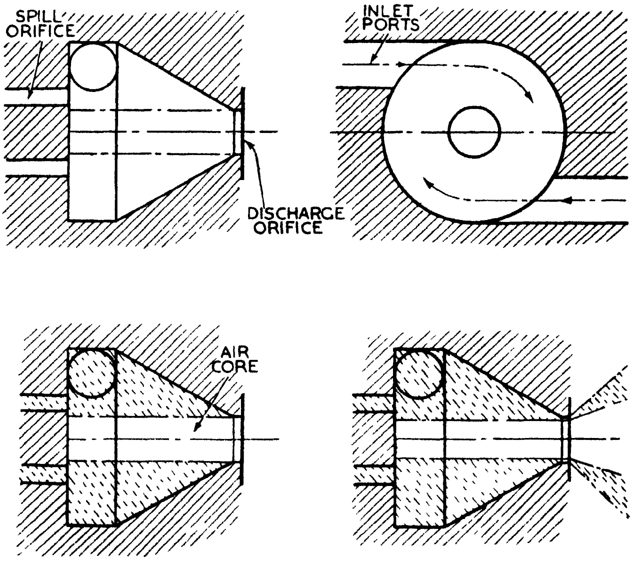
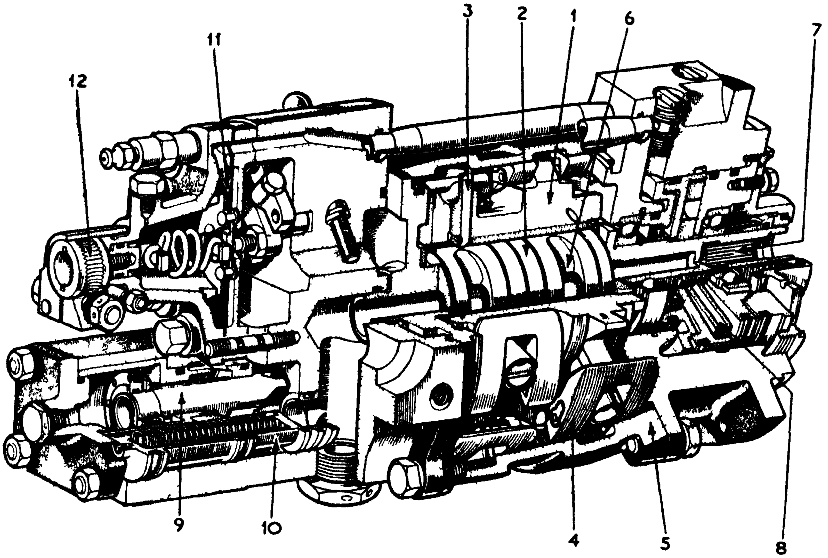
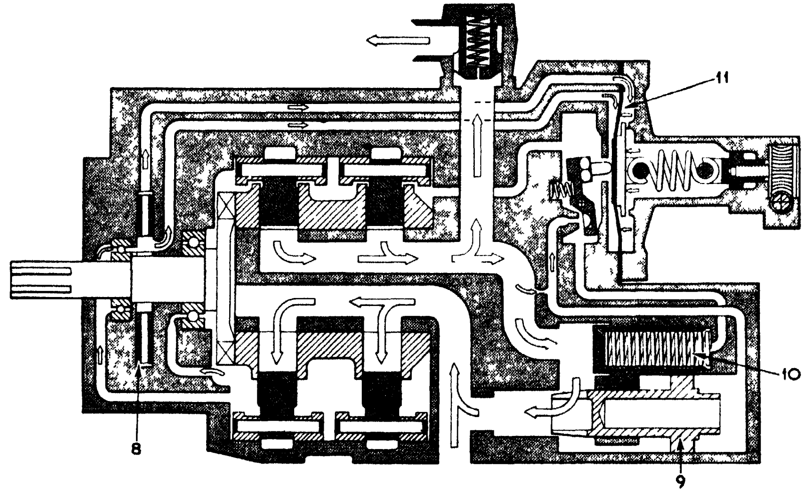


Fig. 7 : 20—Spill burner swirl chamber.



1 Cylinder rotor · 2 Distribution valve 3 Plunger 4 Slipper 5 Track ring 6 Delivery port 7 Quill shaft serrations 8 Centrifuge impeller 9 By-pass valve 10 Trip valve 11 Constant-pressure valve diaphragm 12 Diaphragm loading adjuster

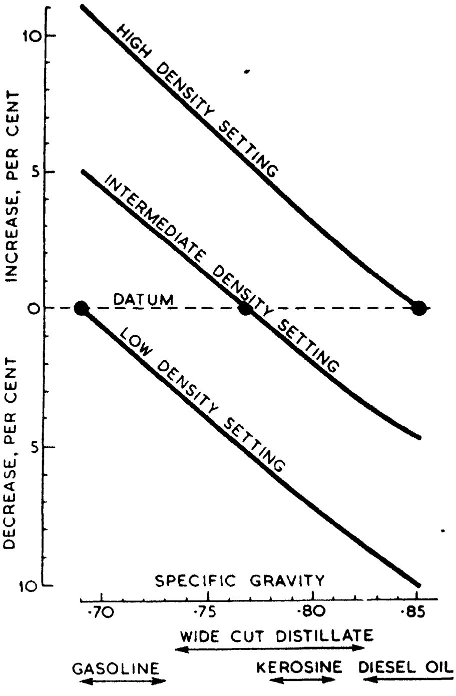
Fig. 7 21 Supply pump with overspeed and universal fuel governors



8 Centrifuge impellers 9 By-pass valve 10 Trip valve 11. Constant-pressure valve diaphragm

Fig. 7 : 22—Fuel flow path in supply pump.

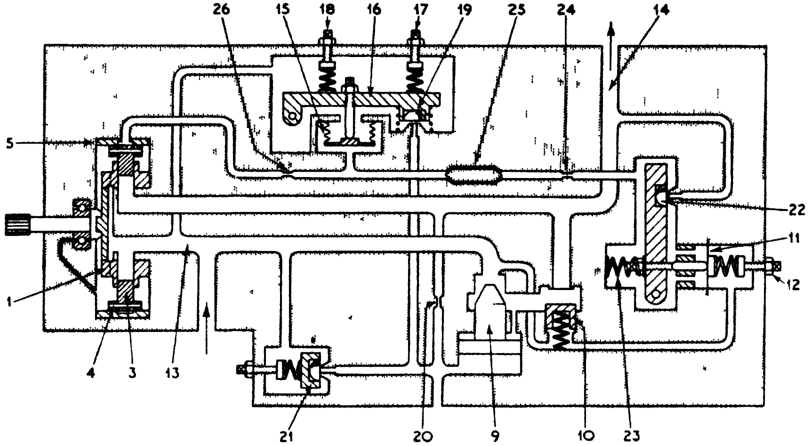
Fig. 7 : 23—Variation of governed speed with different-density fuels.



considerably from batch to batch. Possible departures from a specified governed speed due to differences in fuel density at various settings of a governor are shown in the graph, Fig. 7 : 23.

The operation of both the overspeed governor and the universal fuel governor can be followed from the functional diagram of the supply pump in Fig. 7 : 24. From the inlet passage 13 in the distribution valve, fuel at boost pressure is picked up by the centrifuge impeller (not shown in the diagram) which generates a hydraulic head utilized to give a signal of engine speed and to operate the control of the pump output. The amount of fuel delivered to the burners, and consequently the rotational speed of the turbine unit, is regulated by a by-pass valve 9 in a duct linking the inlet and delivery sides of the pump.

Pressure generated by the centrifuge is tapped off by way of drillings and an external tube to the outside of a bellows assembly 15. An increase in engine speed causes the pressure to rise and the bellows to deflect and contact the tappet of a rocker arm 16. At a preselected speed, determined by coarse and fine adjuster screws, 17 and 18, the rocker is lifted to permit the half-ball valve 19 to open. Fuel then flows from the high-pressure side to the low-pressure side of the pump, causing a pressure drop across the restricted orifice 20. Eventually the pressure drop reaches the value where the load exerted by servo pressure on the full area of the by-pass valve is less than that acting on the annular area on the opposite side and the valve moves outwardly under the differential loading. As a result, fuel is by-passed from the delivery side back to the inlet side and the pump output falls. The valve assumes a stable position according to the operation of the servo system.



1 Cylinder rotor 3 Plunger 4 Slipper 5 Track ring 9 By-pass valve 10 Trip-valve 11 Constant-pressure valve diaphragm 12 Diaphragm loading adjuster 13 Inlet passage 14 Delivery passage 15 Bellows assembly 16 Rocker arm 17 Coarse adjuster 18 Fine adjuster 19 Half-ball valve 20 Restricted orifice 21 Stall valve 22 Constant-pressure half-ball valve 23 Rocker spring 24 Restricted orifice 25 Laminar restrictor 26 Restricted orifice

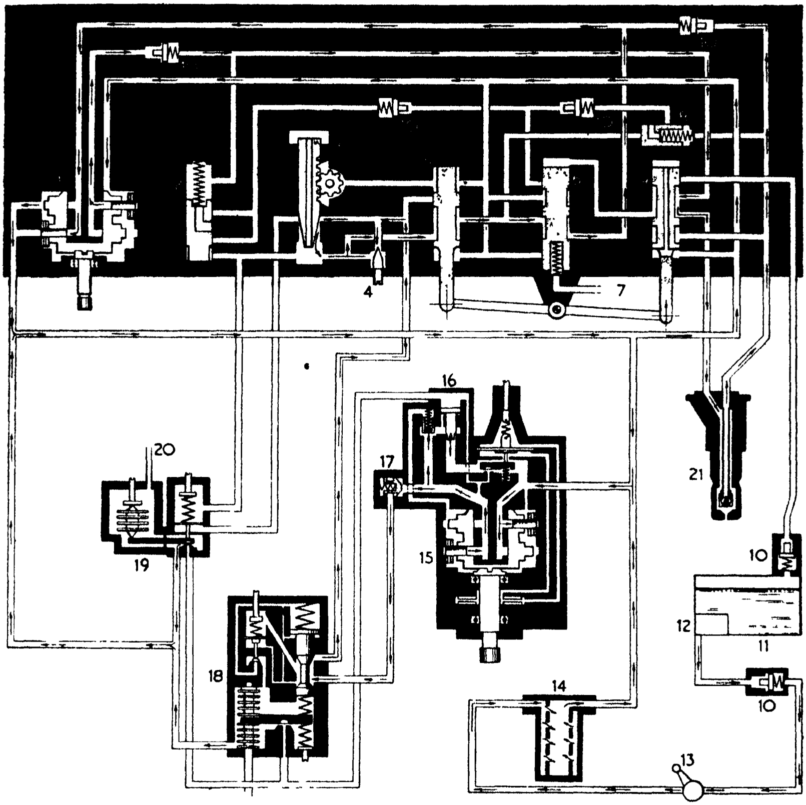
Fig 7 24 –Supply pump functional diagram.

A trip valve 10 in the high-pressure line to the by-pass valve ensures that the initial pressure raised in the pump passes by way of the restricted orifice to pressurize and close the by-pass valve on starting. In the event of the system becoming overloaded, an adjustable stall valve 21, of the half-ball type, is provided to relieve the servo system and thus open the by-pass valve

Imposed on this overspeed governing system is the universal fuel compensating system This furnishes the means whereby servo pressure on the bellows, dependent upon the density of the fuel used, is automatically regulated within the required limits extending from aviation gasoline to light diesel oil.

Fuel from the delivery side of the pump flows past the half-ball valve 22 of the constant-pressure valve assembly to a restrictor line, the back-pressure in which pressurizes the constant-pressure valve diaphragm 11. When this pressure is sufficient to overcome the fuel boost pressure and the adjusted loading spring 12 behind the diaphragm, the rocker arm spring 23 tends to close the half-ball valve. Thus a state of equilibrium is established at a pre-set pressure. Under the influence of this pressure fuel flows through a restricted orifice 24, a laminar flow restrictor 25, and then a second restricted orifice 26 in the centrifuge line.

A fuel of high density and high viscosity has only a small flow through the restrictors and produces only a low pressure while a low-density fuel has a relatively large flow and yields a higher pressure. Since



1. Circulating pump· 2. Isolating valve 3. Throttle 4. Idling adjuster: 5. Shut-off valve No. 2· 6 Circuit control valve: 7. Drain 8. Priming valve 9. Shut-off valve No. 1. 10. Non-return valve. 11. Tank· 12. Tank pump 13. Low-pressure cock 14. Low-pressure filter· 15. Supply pump 16. By-pass valve· 17. Pressurizing valve· 18. Air . fuel ratio control unit: 19. Flow control unit: 20. Nacelle pressure· 21. Spill burner.

Fig. 7 : 25 Typical Dowty spill burner system. Flow shown under normal running conditions.

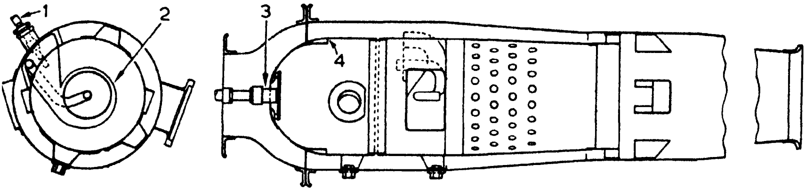
the pressure generated by the centrifuge impeller is high for a high-density fuel, the pressure from the constant-pressure valve by way of the restrictor line has a compensating effect. It follows that the resulting pressure on the overspeed governor bellows is held approximately constant for the same engine speed, whatever fuel is being used.

The Dowty system is fully developed and operational and has successfully concluded a 500 hr test on aviation gasoline. It is standard equipment for the de Havilland Ghost turbojets installed in the Venom night fighters. Components can with advantage be applied to other systems. For example, the pump and governor are fitted on the Sapphire and the

Double Mamba units for fuel supply and control in conjunction with the Armstrong Siddeley vaporizing system. The universal fuel governor can be produced as a separate unit for operation with, or built on to, existing fuel pumps. Pump and governor units are also being produced in America and have been adopted by certain U.S. turbine builders.

ARMSTRONG SIDDELEY VAPORIZER

Unique among British turbine units is the Armstrong Siddeley vaporizing system. When the decision to adopt it was taken it was considered that as the rate of combustion increases with decreasing size of fuel particles, it was logical to secure the smallest possible particles, molecules, by means of vaporization. The heat for vaporization necessarily had to be derived from the combustion of the fuel and consequently the most advantageous location of the vaporizer was in the flame tube. Since vaporization of fuel in the absence of air would result in ' cracking ' of the fuel and give rise to deposits, it was decided to pass air through the vaporizer with the fuel.



1. Four-hole fuel jet: 2. Mixing chamber: 3. Torch igniter: 4. Shoulder ports.

Fig. 7 : 26—A.S.X. chamber with vortex vaporizer.

The first successful vaporizer was produced for the long, small-diameter combustion chambers of the A.S.X experimental turbojet, the precursor of the Python turboprop. Air from the compressor enters at the nose and the majority flows along the annular passage between the casing and the flame tube. The vaporizer is in the form of a lateral duct in the flame tube, terminating in a volute-shaped mixing chamber. Primary air is picked up from the annular passage and fuel is sprayed into the duct by a four-hole injector at a pressure difference of only about 15 $\mathrm{lb/in^2}$ and spread over the wall of the mixing chamber by centrifugal action. The mixture, leaving the mixing chamber by an orifice on the upstream side, enters an envelope of burning gas and the jet travels towards the plate set in the nose of the casing.

From behind this plate a film of air is admitted to flow over the surface of the nose and insulate the metal from the heat of the flame. At the shoulder further air is added, through a series of ports, and combustion is virtually completed when the gases pass downstream over the outer surface of the vaporizer. To ensure that air slightly in excess of combustion requirements is available under conditions of the richest mixtures, additional air is fed through rings of holes in the flame-tube tail member. The gradual admission of air ensures flame stability over a wide range of operating conditions as, although the point in the flow path at which combustion is completed will vary, the charge is never too weak to sustain the flame.

Four rectangular scoops attached to the outer casing beyond the end of the flame tube initiate the mixing of the diluent air with the products of combustion. These direct wedges of cold air towards the centre of the flame and cause an eddying motion.

Ignition is effected in two chambers by means of torch igniters and propagated to other chambers by way of interconnection pipes between adjacent flame tubes. Fine spray starting jets are fitted in chambers not furnished with igniters. As the main fuel feed pressure is low a valve is inserted in the line to raise a pressure difference of 50 lb/in^2 to operate the servo mechanisms of the fuel control system. By means of a solenoid valve actuated by the ignition switch, this system is bled to feed the torch igniters and the starting jets by diverting fuel from the main jets. Thus no fuel extra to that normally fed for idling condition reaches the chamber. In the torch igniter the fuel is fed down the central electrode of the spark plug and issues from a small jet orifice to strike one of the earth electrodes to break into a spray between the spark gap.

Different conditions apply on the Mamba combustion chamber which was required to be as short as possible. To secure a more rapid release of heat it was decided to divide the flame into several portions in order to obtain a greater area of flame front. To this end the vaporizer is formed of four pipes with crook-shaped ends. Fuel and air enter the upstream ends of the pipes and the outlets, turned through 180°, are directed upstream to pass the burning charge over the pipes and thus provide the heat for vaporization. Secondary air is admitted through four slotted nozzles mounted in the diaphragm carrying the vaporizer. They produce fan-shaped air jets, the edges of which strike the vaporizer boss and the flame tube and cause eddies, while the main body of air mixes with the primary charge. Combustion is substantially complete by the time the gases have returned over the vaporizer and the diluent air is injected by way of rectangular ports in the flame tube, which in this instance extends the full length of the chamber casing. A bowed strip of metal is affixed across each port to produce turbulence.

An important factor conducive to reliability in operation is the use of the technique of film cooling. This is applied to the flame tube by means of the so-called ‘ step wall ’ construction. The downstream end of each of the four sections of the tube telescopes, in suitably spaced relationship, into the upstream end of its neighbour. A film of air passing through

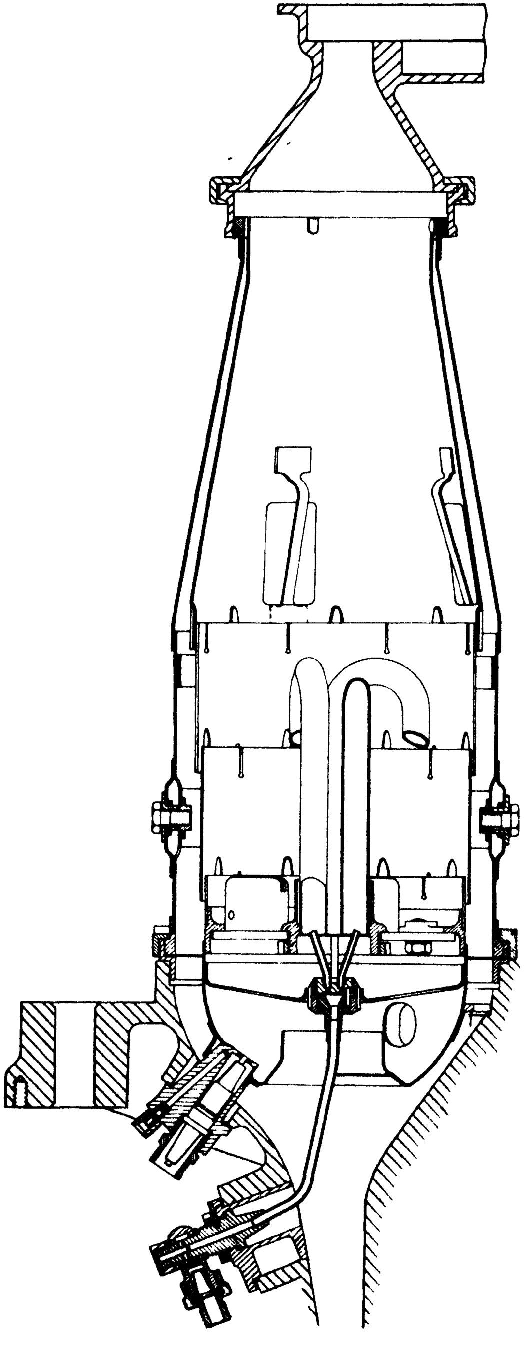


Fig. 7 : 27—Mamba combustion chamber with crook-tube vaporizer.

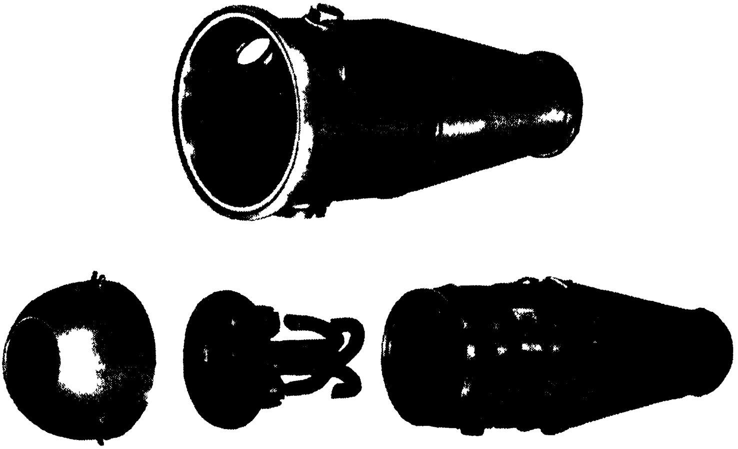


Fig. 7 : 28 -Main components of Mamba combustion chamber.

the intervening space flows parallel to the inner surface of the section and insulates it from the heat of the flame.

In the Mamba ignition occurs in the ante-chamber in which combustion air is collected. The torch igniter, incorporating a standard ignition plug against the central electrode of which fuel is sprayed from a passage in the body, is mounted at an angle to the air entry orifice in the nose. Extremely rapid light-up is obtained as hot gases are fed to the vaporizer and flame passes through the slotted secondary air pipes. To avoid the possibility of damage resulting from prolonged operation it is arranged that the ignition current is cut automatically at a given speed of rotation or after a certain interval of time, whichever is attained first. Igniters

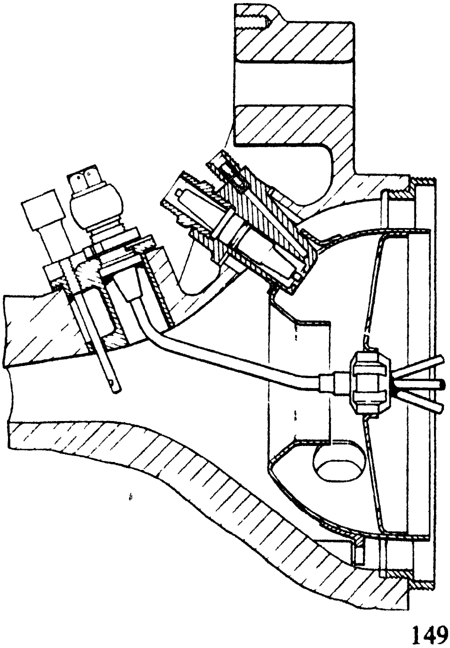


Fig. 7 : 29— Mamba torch igniter.

are fitted to two chambers and fuel sprays to the remainder, as on the A.S.X.

The manufacturers claim a number of advantages for the vaporizing combustion system. It is a low-pressure system and efficiency is not seriously affected by variations of fuel pressure, air pressure, temperature or mixing ratio. Relatively simple fuel supply equipment and plumbing is required. It will operate satisfactorily on a range of fuels varying from aviation petrol to hydrogenated creosote. Armstrong Siddeley chambers have been operated successfully on the test rig using D.T.D. 482B lubricating oil as fuel. It lends itself equally well to individual tubular ‘ cans ’ or to annular chambers.

An example of annular construction occurs on the Viper axial-flow turbojet developing 1,600 lb s.l.s. thrust at 13,400 r.p.m. Twenty-four crook-tube vaporizers are evenly spaced on a mid-located pitch circle with 24 slotted secondary air tubes arranged in inner and outer concentric pitch circles on the forward diaphragm. The chamber is built up of inner and outer rings to produce four parallel sections and nose and tail sections. Each section telescopes into the adjacent section and admits a film of air to insulate the inner surface, as in the Mamba chamber.

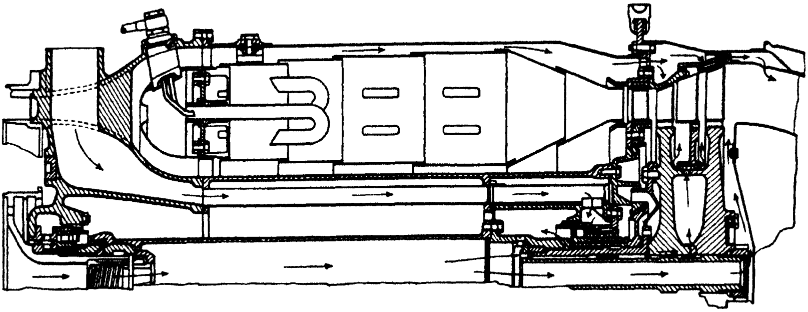


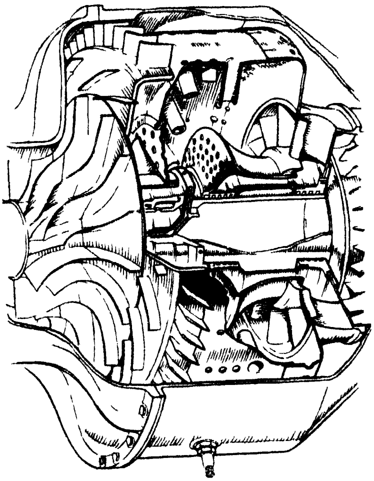
Fig. 7 : 30—Viper annular combustion chamber.

TURBOMÉCA CENTRIFUGAL THROWER

All Turboméca units embody an annular combustion chamber, the inner element of which has radial and axial sections giving an approximate Γ-shaped configuration. Air delivered by the centrifugal compressor past radial and axial diffuser vanes to the outer casing is fed through holes in the walls of the radial section to the primary combustion zone. Secondary or dilution air passes through stub pipes or lipped slots in the outer wall of the axial section.

A gear-type feed pump delivers fuel at a pressure of approximately 45 lb/in^2 to the forward end of the hollow mainshaft. The fuel passes

Fig. 7 : 31—Turboméca combustion chamber and 'centrifugal injection' fuel system.



along the shaft and is flung through radial holes in a thrower disc into the primary combustion zone. At the high speed of rotation the fuel is raised to about 2,000 lb/in^2 pressure, given a velocity sufficient to ensure adequate penetration, and thoroughly atomized.

By this method a high-pressure fuel pump is not required and the need for a large number of calibrated jets is avoided. A uniform distribution of fuel is assured by rotation and even should one or more jets be obstructed, distribution is not affected. As a result good combustion is obtained and the efflux is remarkably free from fumes. Aviation kerosine is the usual fuel, but 100-130 grade petrol or light diesel fuel may be used.

AFTERBURNING

Schemes to burn additional fuel in the jet pipe downstream of the turbine to augment the thrust for take-off or in emergency were features in many early designs and projects, but little was done in the early years to develop the afterburning system. Now that jet aircraft are in regular service a reserve power source to increase the thrust for short periods has become a practical requirement. It is required for take-off, emergency, combat conditions, and possibly for traversing the sonic barrier on ultra-high-speed aircraft.

Afterburning would seem to be a desirable method of thrust augmentation as it imposes no additional stresses on the turbine unit. Admittedly it consumes fuel at a high rate but its specific rate of consumption is less than that of the rocket booster unit. An influential consideration is that it burns the same fuel as the jet unit, thus avoiding complication of installation, of servicing and also of storage on the airfield. Furthermore, some of the rocket motor fuels are relatively sensitive while turbojet fuels are comparatively safe. The additional weight of equipment is trifling. That a 40–50 per cent increment of thrust can be obtained without increasing frontal area, and consequent aircraft drag, is the most important factor.

It is essential that ignition be positive, despite the high velocity of the jet stream. Fuel must be well atomized and well distributed to ensure even heat and velocity gradients in the relatively short length of the jet pipe. Rate of fuel supply must be closely controlled to avoid the possibility of ‘choking’ the exit nozzle, which would adversely affect the turbine. A variable-area propulsion nozzle must be provided to cover normal and reheated conditions.

By courtesy of the Ministry of Supply, reference may be made to a programme of continued investigation and development of full-scale exhaust reheat equipment which was commenced in 1943. Early experiments were concerned with systems having a single axially-arranged pilot chamber for the initiation and stabilization of the flame. Next followed a scheme in which the fuel was injected upstream in the wake of a single hemispherical baffle, with which a thrust increase of 15 per cent was obtained. Eventually all work on central pilot chambers was abandoned in favour of multi-stabilizer systems. In that shown in Fig. 7 : 32 the fuel was injected upstream in a region of high gas velocity to ensure good atomization and the flames were stabilized in the turbulent wakes of five conical baffles. Test results of this system on a Power Jets W2/700 turbojet are given in the accompanying table.

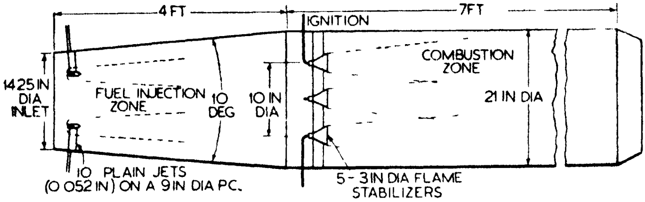


Fig. 7 : 32—Multi-stabilized exhaust reheat system.

All major manufacturers of turbojets have developed afterburning equipment for their high-performance units for military duties. It is commonly arranged that the pressure ratio at the nozzle is un-

TEST RESULTS

Performance of W2/700 turbojet with multi-stabilizer reheat system. Turbine outlet conditions (max.). Velocity. . Mach No. 0·503, Temperature. . 916°K, Static pressure. . 19·98 lb/in^2 (abs).

	Nozzle (*in*)	*Jet pipe length* (*in*)	*Jet pipe dia.* (*in*)	*Thrust* (*lb*)	*Reheat fuel flow* (*lb/hr*)	*Reheat fuel cons.* (*lb/hr/lb*)	*Gross fuel cons.* (*lb/hr/lb*)	*Thrust inc.* (*per cent*)	*Temp. ratio*	*Comb. eff.* ηc
Standard unit. .	12·8	48	14·25	1,910	—	–	1·15		–	—
Equipped unit . .	13·0	84	21·0	1,866	—	– –	—	−2·3	—	—
Reheated unit . .	15·0	84	21·0	2,387	3,280	6·23	2·29	28·3	1·73	83·4
,, ,, . .	16·0	84	21·0	2,525	3,920	5·90	2·42	35·8	1·93	90·4
,, ,, . .	17·0	84	21·0	2,700	5,620	6·70	2·90	45·2	2·20	89·8
,, ,, . .	17·5	84	21·0	2,764	6,970	7·21	3·32	48·6	2·33	88·7

changed so that the turbine is unaffected, and to this end the nozzle area must be increased. Gas velocity, and consequently static thrust for take-off, will increase as the square root of the absolute temperatures, but gas volume will increase in direct ratio to the absolute temperatures. Since nozzle area varies directly with the gas volume but inversely with the gas velocity, it will need to be increased by the same ratio as the gas velocity.

Obviously, an increased nozzle area will be a disadvantage under normal operating conditions without afterburning and, therefore, a variable-area nozzle is required. These devices call for careful design as they must be light in weight, able to withstand the high temperature of the gas, and not susceptible to distortion that might impair smooth operation.

The axially slidable internal bulb or bullet, as used to control the jet characteristics of the Junkers 004 unit (Figs. 12 : 6 and 12 : 7) is less suitable in an afterburner due to the higher temperature of the gas stream. Whereas the exhaust from a turbine is of the order of 930° K, the temperature after reheating may well be twice that figure and problems in relation to cooling, support, distortion, and actuation of the bullet become severe.

Usually resort is made to externally mounted and actuated devices of the clamshell or eyelid type (Fig. 7 : 33), the multi-flap type defining a rectangular orifice, or the segmental or caliper type. Designs are mainly for two-position devices giving normal non-burning and maximum augmented-thrust nozzle areas. That for the Rolls-Royce Avon, with an electro-hydraulically-actuated ‘ eyelid ’ nozzle, is shown in Fig. 7 : 34. The demand is rising, however, for a fully variable nozzle that can be locked in any position. With this a smooth

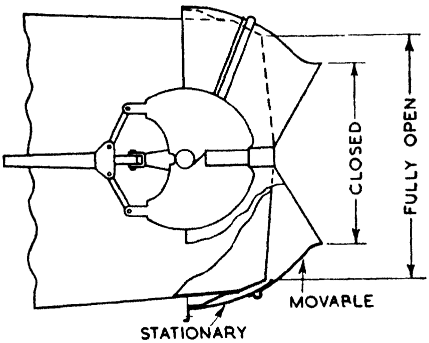


Fig. 7 : 33—Clamshell-type variable nozzle.

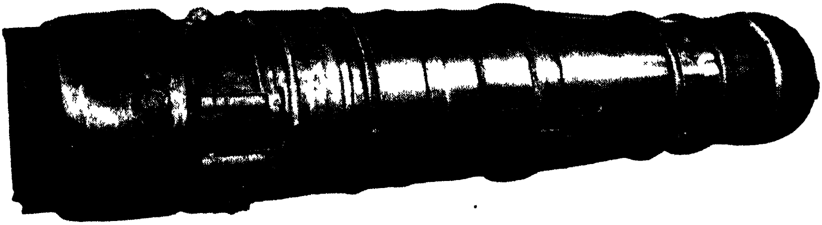


Fig. 7 : 34—Afterburner for Rolls-Royce Avon.

accession of power from normal to maximum thrust is possible and the nozzle can also be used to adjust the jet velocity under non-burning contions and thus improve operating efficiency.

The high specific rate of fuel consumption in an afterburner—a thrust increase of 30 per cent may well double the normal rate of consumption—is likely to overtax the capacity of the main unit pumps. Accordingly, special afterburner feed pump units and the necessary control equipment have been developed. In these a small turbine, rotated by air tapped from the main compressor, drives a centrifugal fuel pump.

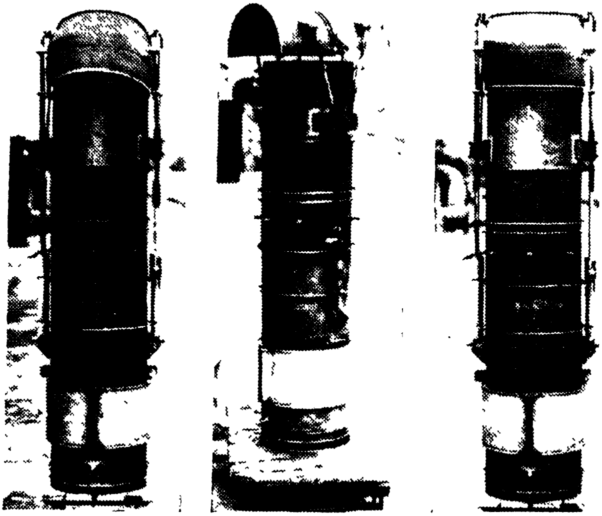


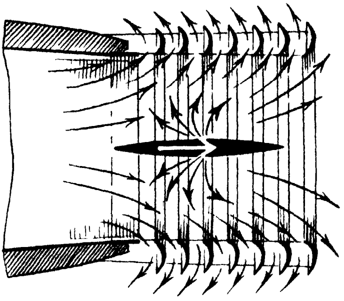
Fig 7 : 35—This early thrust-spoiling device developed by Power Jets would provide a 12 per cent negative thrust for braking.

THRUST SPOILING

Characteristically, jet aircraft are aerodynamically clean and consequently even the small values of thrust generated at idling speeds of the unit are apt to be undesirably effective when coming in to land, particularly in the case of carrier-borne aircraft. A normal approach is made with the unit throttled back and if the landing is baulked the throttle is opened up for a pull-out. In a deck landing this is liable to be attended by some hazard as a few seconds are required to attain full thrust. More desirable would be a method whereby constant r.p.m. could be maintained while the positive thrust was reduced.

Devices to achieve this end have been developed by Power Jets Ltd. and an experimental model is shown in Fig. 7 : 35. This comprises two semi-cylindrical doors hingedly mounted on the exterior of the jet

Fig. 7 : 36—S.N.E.C.M.A. thrust reverser with air-jet deviation.

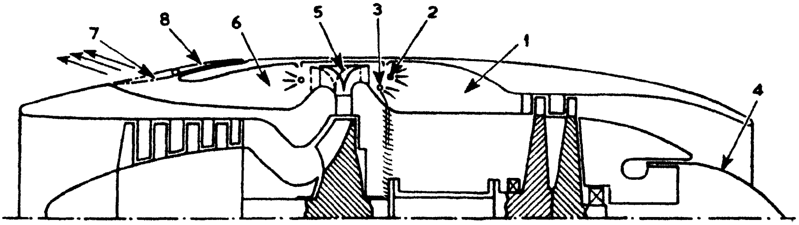


pipe. In the open position the jet is completely unobstructed but when the doors are swung to a closed position, spaced from the end of the pipe, the jet is deflected to produce a negative reaction. Negative thrust to a value of about 12 per cent of the unimpeded positive thrust can be obtained with this device. Intermediate positions can be used if required to vary the thrust.

In France the S.N.E.C.M.A. organization has produced a jet braking device termed a ' thrust deviator.' Essentially it comprises a series of channel section rings mounted around the propulsion nozzle. The jet stream is diverted into these rings and deflected forwards. To avoid damage to the structure, and also to prevent pitching movements likely to affect control of the aircraft, the gas stream is deflected only over sectors of the nozzle area.

During normal running the jet stream passes through the rings with little loss of energy. In an early model diversion was effected by variable incidence vanes which induced rotation and delivered the jet stream into the rings by centrifugal force. A later version dispersed the stream into the rings by means of a jet of compressed air from a nozzle in the jet orifice. Demonstrated on a Vampire at Le Bourget it effectively braked a landing and even enabled the aircraft to be taxied backwards.

Another French scheme, patented by the Office National d'Etudes et de Recherches Aéronautiques (O.N.E.R.A.), diverts air from the compressor to one or more auxiliary combustion chambers delivering gas to forwardly directed braking nozzles. In a diagrammatic embodiment of the invention, Fig. 7 : 37, a l-p axial compressor and a h-p centri-



1. Annular combustion chamber: 2. Fuel burner: 3. Water atomizer: 4. Adjustable inner cone: 5. Toroidal deflector: 6. Braking combustion chamber: 7. Braking nozzle: 8. Nozzle flap.

Fig. 7 : 37—Diagram of O.N.E.R.A. jet braking system.

fugal compressor deliver air to an annular combustion chamber 1 furnished with the usual fuel burners 2 and also water atomizing jets 3. A two-stage turbine drives the compressors, and the propulsion nozzle area is variable by means of a slidably mounted ogival inner cone 4.

The air stream from the centrifugal compressor can be divided by means of an axially displaceable toroidal deflector 5 and some portion—from 50 to 80 per cent—diverted to the braking combustion chamber or chambers 6. Here fuel is burned and the resulting gas stream is discharged forwardly through nozzle or nozzles 7 which, in normal flight, are closed by flaps 8. These nozzles should be so disposed that the issuing jets do not impair visibility for the pilot, that no part of the aircraft has to fly through the hot gas and the hot gas is not taken into the compressor intake.

Preferably they should diverge laterally and upwardly, and in order to group the chambers in the upper part of the unit the air from the compressor may be collected in a scroll running over the whole periphery. Since these chambers are subject to only low, static stresses and are operative for only short periods, they may be made of relatively low-strength refractory material.

In some units it may be of advantage to increase the pressure drop through the turbine by varying the area of the propulsion nozzle. In such cases water may be injected in the main combustion chamber to compensate for the reduction of the rate of flow through the chamber due to the diversion of air to the braking combustion chambers.

Operationally it is claimed that approach speed can be reduced, landing run can be shortened and the reaction of the upwardly directed braking jets will tend to hold the aircraft to the ground on landing. In the event of a baulked landing or a wave-off from a carrier, the braking chambers can be cut out and the rotor accelerated to restore rapidly maximum forward thrust. In the air the braking jets may be used to lower the

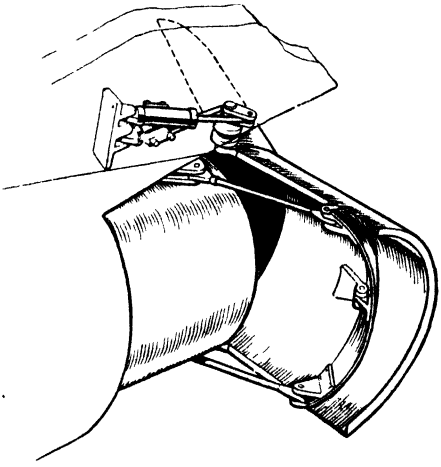


Fig. 7 : 38—The Boeing thrust deflector has a pair of power-actuated semi-cylindrical doors.

aircraft speed and thus make possible a reduction of the radius of turn in manœuvres.

Numerous other devices are under development. In America a thrust-reverser (Fig. 7 : 38), somewhat on the lines of the earlier Power Jets unit, has been announced by the Boeing company. The N.A.C.A. is developing a system in which channel-section members arranged on the axis of the jet stream are displaced laterally, alternate elements to the right and to the left, to deflect the gases through orifices in the tail pipe. Shutters actuated simultaneously with the channel deflectors close the orifices during normal operation.

STARTING EQUIPMENT

Rapid increase in the size and power output of turbine units brought in its train insistent demands for starting equipment more powerful than the conventional electric motor and storage battery type. A gas turbine requires a heavy peak current momentarily for initial rotation and a substantially sustained current to bring it up to firing and idling speeds At the peak 30 to 40 kW is necessary for a turbine of modest size while 60 to 70 kW may be required for the largest units. Such demands cannot be met efficiently by storage batteries of normal type. The problem becomes more acute in the multi-unit installations on large aircraft.

For ground starting, particularly in the case of civil aircraft, recourse is had to mobile generator sets, comprising a diesel engine driving a special D.C. generator. Peak outputs of 90 to 95 kW at 112 V or 50 kW at 28 V are available or approximately 50 kW and 20 kW continuously at the respective voltages for servicing and testing purposes. Typical sets are produced by Auto Diesels, Ltd., Uxbridge, and Petbow, Ltd., Sandwich.

A completely self-contained system, however, is desirable for military aircraft and, of course, is essential for restarting in the air. This aspect of the problem was realized from the earliest days and the Junkers Jumo turbojet was started by a Riedel flat twin, two-stroke, petrol engine housed in the air intake bullet (Fig. 12 : 6). With a bore of 70 mm and stroke of 35 mm, giving a swept capacity of 270 cm^3, it developed 10 h.p. at the maximum speed of 10,000 r.p.m. Weighing only 36 lb, it was started electrically from the cockpit, but by cable and pulley when on the ground.

Such a unit would be quite inadequate for modern large turbojets and a solution of the problem has been provided by the turbo-starter operated by a cordite cartridge. Typical of these is the Rotax unit, which was the first in the world to be officially approved and passed into regular production. Fig. 7 : 39 shows sectioned the CT.0101 starter developed for the de Havilland Ghost turbojet, which it can bring up to idling speed in 10 sec instead of the 50 sec commonly required with electric starters. Two cartridges are accommodated, permitting a second attempt after 20 sec should the first fail to effect a start. It is rated to deliver 137 b.h.p. at an output shaft speed of 8,250 r.p.m. and duration of operation is 3·2 sec. Total weight is 60 lb.

To operate, the pilot first actuates a switching device to select the cartridge to be fired. This automatically arranges for the circuit of the alternate cartridge to be established after the first operation but prevents it being fired until an interval of 20 sec has elapsed. Depression of a push button fires the cartridge which normally burns about 3 sec and the gas released by combustion, at a maximum temperature of 1,750° C and pressure of 1,200 lb/in^2, passes to a chamber fitted with a bursting-type safety disc. From this chamber the gas passes through four tangential nozzles, and coincident ports in the overspeed cut-off plate, to impinge on the impulse-type rotor blades. The efflux is exhausted to atmosphere. From the turbine rotor the drive is through a clutch and an epicyclic gear giving a reduction of 5 : 1.

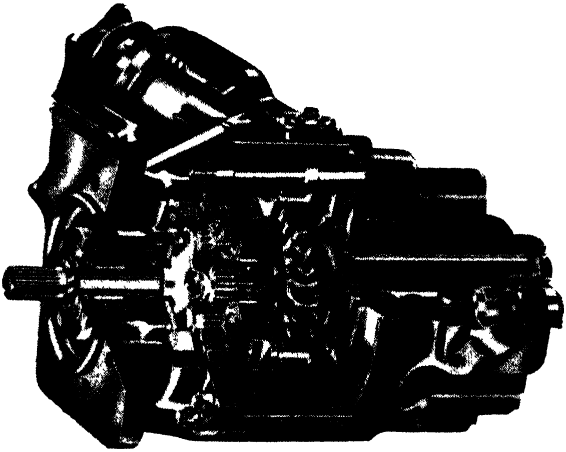


Fig. 7 : 39—Rotax cartridge starter for de Havilland Ghost turbojet

Acceleration of the rotor is rapid and initially the overload clutch, set to transmit 160 lb/ft, is slipped. At the moment of engagement the drive shaft is stationary. As the inertia of the main power unit is overcome, slip is reduced and the starter shaft in 1 sec attains a speed of about 2,500 r.p.m. and reaches about 7,500 r.p.m. in 3 sec. Should the drive to the power unit be broken, or the second cartridge be fired while the power unit is rotating at relatively high speed, serious overspeeding of the starter turbine could occur. To prevent this a centrifugal clutching device operates at 50,000 r.p.m. to move the cut-off plate through 30° and divert the gases away from the rotor and direct them to the exhaust.

Under certain operational conditions the possibility of only two starts per loading could constitute a handicap. To circumvent this a change from the solid-propellant cartridge to a liquid propellant could furnish a number of starts only limited by tankage capacity. The Rotax starter has been developed for operation by liquid propellant and versions for both Avon and Sapphire turbojets are available.

The mono-propellant used is iso-propyl nitrate, known commercially as ' Catalene B.' It is slightly heavier than water, is officially classified as non-explosive, and as regards the safety aspect is regarded as similar to gasoline. Due to the presence of free nitric acid it corrodes ordinary steel but stainless steel and aluminium are unaffected. Flash point is 70° C, while the addition of glycol—5 cm^3 to 1 gal—depresses the freezing point to —40° C. When atomized and ignited it decomposes to a mixture of gases, predominantly hydrogen and carbon monoxide, at a temperature of 1,100° C.

Designed so that it can be mounted inside an intake bullet, the starter comprises a turbine with shock clutch, reduction gear and over-speed cut-off as in the cartridge starter, while the breech block is replaced by a reaction chamber. Additionally, a centrifugal governor driven by bevel gear from the output shaft cuts off the motor driving the propellant pump and also the ignition when the main power unit reaches light-up speed. This 24 V battery-operated motor also drives a small centrifugal fan used to blow clear the reaction chamber between shots.

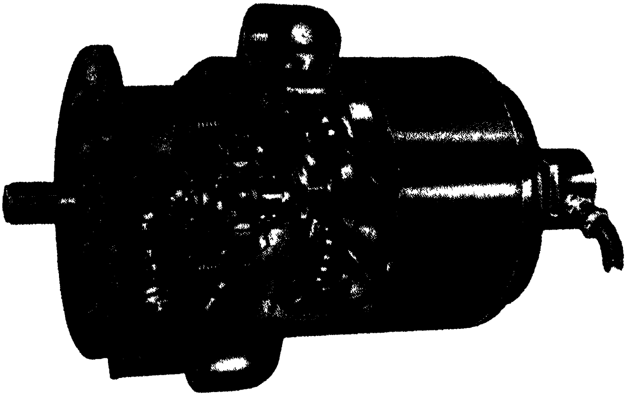


Fig. 7 : 40—Rotax liquid-propellant turbo-starter.

Operation is by push button with a standard time switch unit giving an automatic control sequence. A power unit can be started and brought up to idling speed in 10 to 15 sec and a second shot to a rotating main unit can be made 15 sec after a first failure. To avoid the hazard of self-ignition, all propellant lines in regions of potentially high temperature, such as a de-iced bullet nose and the starter itself, are automatically drained while other lines are maintained fully primed to prevent air locks and the risk of detonation.

Whatever the method used to rotate a turbine power unit the initial ignition of the atomized fuel for a re-light in the air presents special problems which become greater as altitude and speed are increased. The ordinary booster coil and igniter plug system is unsatisfactory for altitudes above 25,000 ft. It has been found necessary to store energy in a condenser and then discharge it across a surface-discharge type of plug to ensure ignition at up to 55,000 ft and with high air mass flow.

In the Rotax NB.25 high-energy ignition unit, operated from a 24 V D.C. supply, a conventional booster coil is used to charge, through half-wave selenium rectifiers, a 6 μF condenser to 2,000 volts. A rising voltage across the condenser will traverse a sealed discharge gap and a small inductance extends the duration of the spark to about 100 microsec. More than 60 sparks per minute are produced, which is more than adequate to ensure ignition in the severely limited optimum period during the 2 to 5 sec run-up time when a cartridge-type starter is employed.

Two of the largest manufacturers of British turbine starters are Plessey and B.T.H. The former has developed both solid-fuel cartridge starters and liquid monopropellant starters, both being installed as self-contained units needing no supply of external power whatever. The B.T.H. single-breech cartridge starter has been made in very large numbers for most early Avons and it has been succeeded by a triple-breech unit in which three cartridges are mounted on the nose of the engine, faired beneath a cylindrical casing.

In the U.S.A., in particular, great attention has been paid to developing air-turbine starters. These units consist of small turbines, of either the axial- or radial-flow type, impelled by an externally supplied flow of hot, compressed air. Rotating at speeds up to 80,000 r.p.m., these small turbines can develop several hundred horsepower and, when suitably geared down, provide a very high starting torque. In most modern American aircraft powered by multiple gas turbines, ducting within the aircraft can feed compressor-bleed air from one engine to the others so that, once an engine has been started by its air-turbine starter from a ground supply, each engine can start its neighbour.

One of the major manufacturers of air-turbine starters is AiResearch, who also make self-contained gas-turbine compressor sets with which an aircraft can be provided with a supply of compressed air independently of ground supplies. Hamilton Standard also make pneumatic starters, a recent model having a 7 in diameter wheel rotating at 45,000 r.p.m. on air at temperatures up to 700° F, and developing over 100 h.p. This unit, which is fitted to the J57-powered McDonnell F-101, is stated to be much lighter than corresponding electric starters.

CHAPTER 8

Metallurgical Problems

CONSTANT-PRESSURE gas turbines function on a continuous combustion process. The mean operating temperature of the working medium, therefore, is higher than that obtaining in the piston engine which works on an intermittent combustion cycle. Moreover, the high velocity of the gases flowing through a turbine unit is liable to create problems of erosion. The gas turbine, in both static and rotating parts, provides most arduous instances of high temperature operation. Conditions are severe and cannot always be anticipated since, in some components, calculated conditions are combined with unknown and varying effects of temperature gradient.

The main effect of stress at high temperatures is continuous deformation or ' creep ', which will eventually lead to the failure of the part by loss of dimensions or even fracture. A typical creep curve, Fig 8 : 1, shows that the course of strain is not consistent. There is a primary stage in which the initial rate of creep decreases rapidly, a secondary stage during which the rate decreases slowly or may be substantially constant, and a tertiary stage when the rate increases rapidly and leads eventually to fracture. Present knowledge of conditions during the tertiary stage prior to fracture is incomplete and it is wise to limit the working life of a component to the determinable period of the secondary stage.

Resistance to both corrosion and oxidation should be of a high order since under conditions of tensile stress they influence the time to the onset of tertiary creep. Furthermore, reduction of metal thickness due to inadequate resistance results in loss of load-carrying capacity. Thermal expansion is also important and components with normally only little stress must possess adequate strength to carry additional stresses imposed by temperature gradients. Thermal conductivity may be a factor of some influence where conduction of heat through the metal, relative to that through the oxide, is of importance.

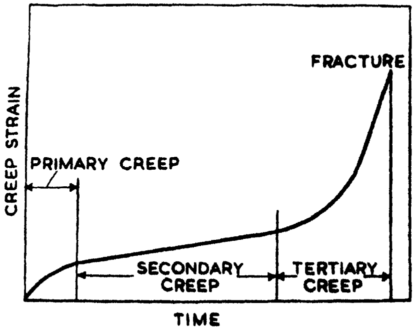


Fig. 8 : 1—Typical creep curve.

DISC FORGINGS

By reason of the high speeds of rotation the turbine disc is subject to heavy stressing. Centrifugal force, tending to burst the disc, varies as the square of the velocity, it will be remembered. Thus the centrifugal loading of a given disc rotated at 15,000 r.p.m. will not be 1·5 times but 2·25 times the loading at a speed of 10,000 r.p.m. The peripheral velocity of a 12 in-diameter disc running at 15,000 r.p.m. is 47,000 ft/min (535 m.p.h.) whilst at the tip of the blades, say 18 in-diameter, the velocity is 70,650 ft/min (803 m.p.h.). The design of discs rotating at high speeds has for years been the subject of intensive study and mathematical analysis, particularly in regard to the rotors of steam turbines. Shape and dimensions can be determined to meet requirements with relative precision but in the case of the gas turbine a new factor is introduced by the higher temperature of the working medium.

Many grades of steel possess adequate strength in the lower temperature ranges but cannot maintain it at elevated temperatures. In operation there is a substantial difference in temperature between the hub and peripheral portions of the disc, say 300–350° C, which produces a thermal stress in addition to the centrifugal loading.

Development of gas turbines for industrial purposes, for which a long working life was essential, was delayed for many years by lack of metals capable of withstanding high stress at temperatures approaching red heat. The prospects of commercial application of the gas turbine were rather vague and consequently research was only of a generalized character.

A tremendous stimulus was received when, immediately prior to the commencement of the war, the practical realization of jet propulsion and the aircraft turbine enabled well-defined conditions of stress, temperature, creep and working life to be specified. Thereafter, by drawing upon the accumulated data of the research in the previous thirty years, progress was rapid. A graph, Fig. 8 : 2, shows the improvement in heat-resisting steels produced by the Sheffield firm of William Jessop and Sons Ltd., in recent years.

In the early days of jet propulsion units the majority of turbine discs used by British manufacturers were of steel made by Jessop. This position stemmed from the initiation in 1936 of a research programme aimed to meet possible future needs for materials possessing high strength at elevated temperatures. The firm's G.2 steel, which was extensively used for the exhaust valves of high-powered aircraft engines before the war, formed an intermediate step to the G.18B steel developed by D. A. Oliver and G. T. Harris, then respectively director and manager of the Jessop research team. This steel was literally ' produced to order ' for the early Whittle jet turbines and has been used from the prototype Welland units to the present time.

Many problems had to be solved in regard to melting, forging, machining and welding before the steel could be placed in commercial production. Obviously, a steel possessing high hot-strength is by nature resistant to forging or other hot deformation, but the as-cast structure must be broken down if high strength, toughness and, in particular, uniformity are to be obtained. A single steel cannot, of course, fulfil the requirements of all the different turbine components to the best advantage and accordingly a range of materials has been developed to meet differing operating conditions.

It soon became evident, however, that for turbine discs it was necessary to bridge the gap between the ferritic and the austenitic steels. Designs involving temperatures of 550–650° C posed a difficult problem

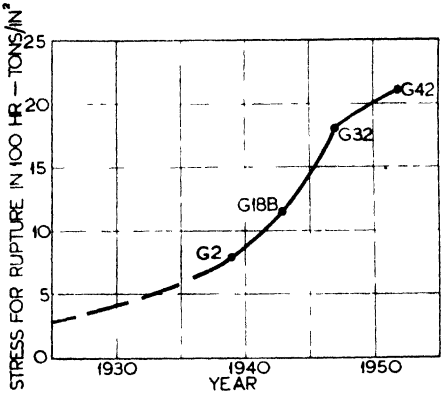


Fig. 8 : 2—Creep strength of Jessop steels at 750°C.

At temperatures of this order, ferritic steels rapidly lose their strength and resistance to scale but austenitic steels do not show to advantage until higher temperatures are reached. In the range of Jessop steels, H.40 was of use at an early stage as it provided a creep strength at 600° C comparable with many austenitic steels although it had the disadvantage of a low scaling resistance. The latter development H.46 had a slightly superior creep strength but its improved scaling resistance permitted its use at temperatures up to 650° C while still retaining the desired qualities, particularly of high proof stress and low thermal expansion, characterizing a ferritic steel. It is steels of this kind that have now largely replaced the austenitic steels for turbine discs. In Fig. 8 : 3 it will be seen that for H.40 the limiting temperature for creep strength exceeds that for scale by a considerable margin. This contrasts sharply with the austenitic steel R.23 which has a lower limit for creep but a very much higher one for scaling resistance.

The chart also shows the advances made by G.18B and the later

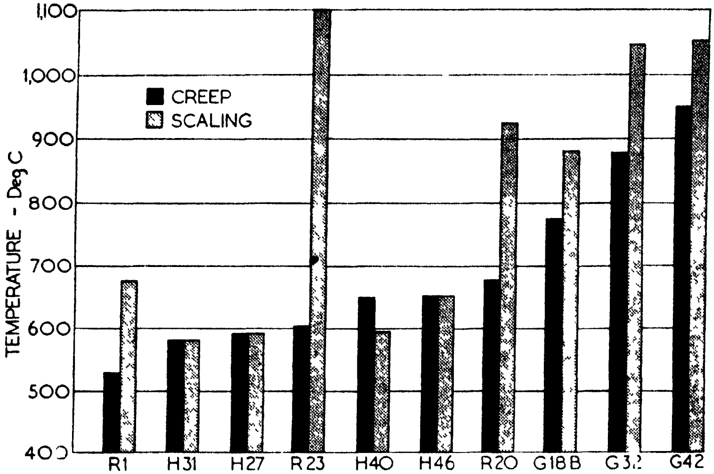


Fig. 8 : 3—Limiting operating temperatures for various Jessop steels.

G.42 over the earlier austenitic steels R.20 and R.23. Chemical compositions of typical steels for turbine discs are :

CHEMICAL COMPOSITIONS

Steel	C	Mn	Si	Ni	Cr	Mo	V	W	Co	Nb
H.40	0·23	0·3	0·45	0·3	2·7	0·5	0·75	0·5	—	—
R.20	0·15	0·8	0·3	14·0	19·0	—	—	—	—	1·7
G18B	0·4	0·8	1·0	13·0	13·0	13·0	2·0	—	10·0	3·0
G.32	0·3	0·8	0·5	12·5	19·0	2·0	2·8	—	45·0	1·3

It was found that a ferritic steel, possessing higher strength at the working temperature of the central portion of the disc and a lower coefficient of expansion, enabled a disc of relatively lighter weight and consequent lower moment of inertia to be employed.

To reduce the thermal stressing of the disc and the transfer of heat from the disc to the shaft and bearings it is usual to employ some form of cooling. Designs have appeared and units have been built having a water-cooled disc and water-cooled bearings but this system is somewhat anomalous in a power unit employing such vast quantities of air and it is usual to cool the rotor by a continuous blast of air passing over the face of the disc from hub to rim. Rolls-Royce Derwent and de Havilland Goblin are examples in which cooling air is applied respectively to one face and both faces of the disc.

Thermal stresses at different temperatures can be estimated from the physical characteristics set out in the following tables :

PHYSICAL PROPERTIES

Steel	H.31	H.40	H.46	R.22	R.20	G.18B	G.32	G.38	G.39	G.42
Specific gravity	7·85	7·84	7·75	7·9	7·95	8·13	8·26		8·5	8·24
Thermal conductivity (g.cal/cm^2/sec/cm/°C)										
100°C	0·100	0·077	0·065		0·036	0·032	0·035			
300°C	0·100	0·081	0·066		0·044	0·041	0·043			
500°C	0·086	0·077	0·066		0·052	0·049	0·051			
700°C	0·071	0·069	0·066		0·064	0·057	0·058			
Mean coefficient of linear thermal expansion ($\times 10^{-6}$ per °C)										
20–100°C ..	12·3	12·4	9·3	14·6	17·0	15 6	14·8	16·5	12 9	11·3
20–200°C ..	12·6	12·4	10·9	15·1	17·2	15·8	15·0	16·5	13·1	13·9
20–300°C .	13·2	12·5	11·3	16·4	17·4	16·5	15·1	16·7	13·2	14·6
20–400°C ..	13·7	12·9	11·5	16·7	17·6	16·9	15·2	17·1	13·4	15·0
20–500°C ..	14·1	13·2	12·0	16·8	17·9	17·1	15·3	17·2	13·7	15·1
20–600°C ..	14·4	13 6	12·1	17·2	18·6	17·3	15·9	17·6	14·2	16·1
20–700°C ..	14·6	13·9	12·1	17·5	18·8	17·7	16·4	18·1	14·7	16·3
20–800°C ..	–	14·0	12·2	17·5	19·2	18·0	16·8	18·4	15·2	17·0
20–900°C ..	—		–	17·7	19·4	18·3	17·2	19·0	15·5	17·3
Modulus of elasticity (tons/in^2 $\times 10^3$)										
20°C	12·7	13·6	14·1	12·7	12·5	13·4	14·5	12·9		
400°C	11·6	12 1	12·4	10·8	10·2	11·6	12·9	10·8		
500°C	10·5	11·5	11·6	10·3	9·4	11·1	12·5	10·3		
600°C	8·1	10·6	10·5	9·9	8·5	10 5	12·0	9·7		
700°C	–	9 4	9·5	9·4	7·6	10·0	11·3	9·2		
800°C	—	7·9	--	8·9	—	9·5	10·6	8·7		

TENSILE PROPERTIES AT ROOM TEMPERATURE

Steel	*HEAT TREATMENT*	0·1 *per cent Proof Stress* (*tons/in*^{2})	*Ultimate Stress* (*tons/in*^{2})	*Elongation* (*per cent*) $L = 4\sqrt{A}$	*Reduction of Area* (*per cent*)
H.31	Hardened and tempered	43	58	20	40
H.40	„ „ „	50	60	18	45
H.46	„ „ „	46	60	20	50
R.22	Air-cooled	27	50	21	30
R.20	„	15	40	45	60
G.18B	Solution-treated..	14	46	40	45
G.18B	Solution-treated and aged	18	48	25	22
G.18B	Warm-worked	25	48	12	15
G.32	Solution-treated and aged	38	64	10	10
G.38	Solution-treated and warm-worked ..	24	41·5	48	52
G.39	As cast	—	33	5	—
G.42	OQ 1280°C, W.W. 6·8 per cent ..	43	55	21	22

The table of the tensile properties of various steels at room temperature also shows, in the case of G.18B, the effects of different heat treatments. The proof stress can be taken as the limit of stress which may be permitted at the relatively cool centre of a turbine disc.

Creep strength, however, is the most important single property of a steel intended for high temperature service in a gas turbine. Fig. 8 : 4 is drawn on a basis of the stress to produce a creep rupture in 1,000 hr.

Discs are forged in G.18B and H.46 to over 30 in diameter and with boss thickness up to 7 in. Certain types of discs have been forged with a short stub-shaft about 5 in diameter and 10 in long, the disc being 22 in diameter and about 4 in thick. The firm states that there is no limitation within reason to the length of shaft that can be produced by this method, except facilities for obtaining a particular heat treatment.

Another possibility of interest to designers of axial-flow gas turbine units is that of producing solid rotor forgings in G.18B steel, with drum diameters up to 18 in, and an axial length of 3 ft, increased by integral extension shafts to 7 ft. Special methods of heat-treatment have been developed for such large components and the manufacturers report that creep tests have given results within 15 per cent of the figure claimed for smaller components under precise laboratory control.

For such highly stressed components, meticulous testing is of paramount importance. In addition to normal inspection, special creep tests to verify that the resistance to deformation at high temperatures is adequate and special examinations to ensure complete freedom from any internal defects are necessary.

Large rotors for marine, locomotive and industrial gas turbines are forged in separate disc and shaft sections which are individually machined and tested before welding into a composite whole. For flaw detection X-rays or gamma rays from a radio-active source are used and also an

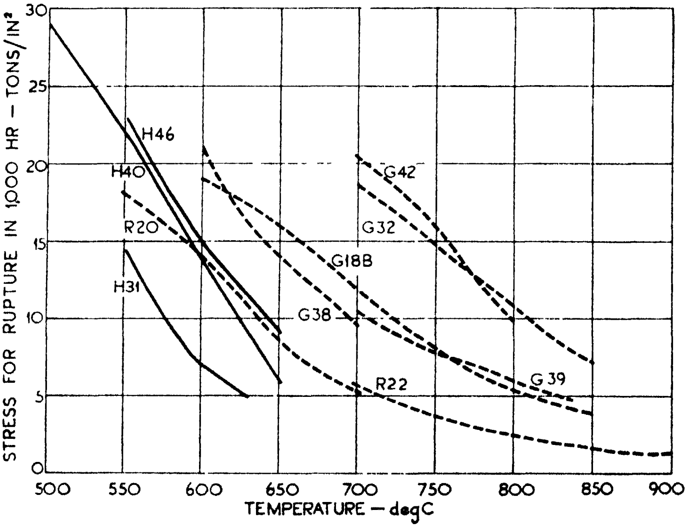


Fig. 8 : 4—Creep properties of three ferritic and seven austenitic steels.

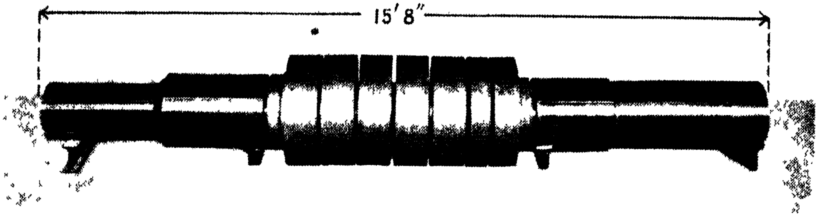


Fig. 8 : 5—Large rotor sections prior to welding.

electronic method employing ultrasonic waves. This latter technique, developed in Britain by H. Hughes & Sons, Ltd., found one of its first major applications in proving the internal soundness of gas turbine rotor forgings.

TURBINE BLADES

The blades of the turbine operate under even more arduous conditions than the disc. Necessarily of light section, they are exposed to the highest temperature and subjected to heavy centrifugal loading due to rotation at high angular velocity, to a bending stress arising from the pressure of the impinging gas and to bending and torsional oscillation. Considering a hypothetical turbine producing 5,000 h.p., this would probably have about 55 blades approximately 3 in long, each of which must transmit continuously about 90 h.p. At the high rotational speeds used, the tensile stress due to centrifugal force will exceed the bending stress and may be of the order of 16 tons/in^2.

For maximum efficiency, the gas should enter the turbine at a high temperature which, as indicated, is limited to the capacity of the materials used in the component parts to withstand the effects of such temperature. Dr. Meyer has stated that on a compressor-turbine unit having an overall efficiency of 76 per cent, to increase the gas temperature at the turbine inlet by 20 per cent from 540° C to 650° C would raise the cycle efficiency from 18 to 23 per cent, representing an improvement of 28 per cent.

Ordinary grades of steel are quite inadequate for turbine blades. In general the special steels are complex alloys containing high percentages of chromium and nickel and smaller additions of other elements such as silicon, tungsten or molybdenum. A steel having a chromium content of 12–14 per cent is superior to ordinary steels as regards resistance to scaling and the maintenance of mechanical strength up to moderately high temperatures. Above 650° C, however, its strength decreases rapidly and at 850° C the non-scaling characteristic is lost. The addition of silicon, up to about 3 per cent, has been found to produce marked improvement in the scaling resistance and also to impart some improvement in strength at high temperatures.

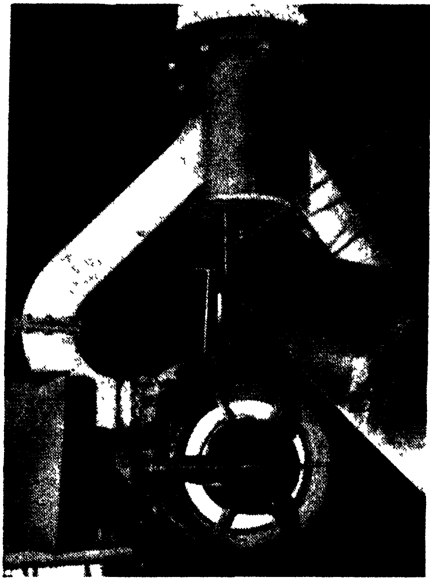


Fig. 8 : 6—A Brown Boveri exhaust turbo blower set on the test bed, running continuously at 30,000 r.p.m. with a gas temperature of 1,000°C.

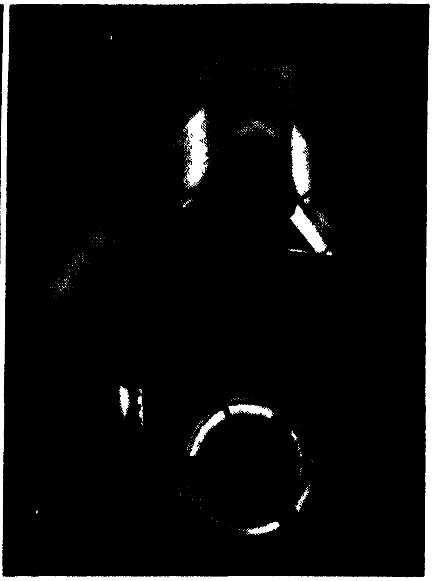


Fig. 8 : 7—The night photograph of the charging set in Fig. 8 : 6 shows the turbine operating at full load with the rotor and guide blades glowing at a red heat.

Exhaust gas turbines for driving the supercharging blowers of diesel engines operate at about 600° C but on petrol aircraft engines the exhaust temperature may reach 1,000° C. These conditions are being satisfactorily met with existing steels. As long ago as 1933, the late Sir Robert Hadfield was able to report that prolonged research by his firm in collaboration with S.A. de Commentry Fourchambault et Decazeville, of Imphy, France, had resulted in the evolution of a high-percentage nickel-chromium steel possessing exceptional strength and non-scaling properties at high temperatures. Designated 'Era/ATV' steel, it was being used at bright red heat and under high centrifugal and other stresses in exhaust gas turbines working at a temperature of 800° C to 930° C and at speeds ranging from 30,000 r.p.m. to 50,000 r.p.m.

For a number of years the Brown-Boveri Company in Switzerland has had a large test bed for testing exhaust gas turbine-driven blowers for petrol aircraft engines. The test equipment is designed to withstand temperatures up to 1,100° C. Figs. 8 : 6 and 8 : 7 show a charging set of their manufacture running with a gas admission temperature of 1,000° C at the designed full speed of 30,000 r.p.m. In Fig. 8 : 6 the equipment is fully illuminated and on the left shows the end of the duct discharging air to simulate the relative wind during flight, and on the right, the collecting funnel for the convenient evacuation of the turbine effluent. The night picture of the same set in operation was taken by the light radiated from the exhaust gas piping and the turbine. The ring of light at the turbine is actually produced by the stationary guide blades which are heated to a bright red. The rotor blades, at a somewhat duller red, cannot be distinguished owing to the high speed of rotation.

Both Jessop G.18B and G.19 steels have been used for nozzle guide vanes and rotor blades, and Fig. 8 : 8 shows typical components in both the stamped and finished conditions. These are drop-forged from flawless ground bar and after receiving the necessary heat treatment are brought to final form by machining, grinding and polishing. Jessops later developed other turbine blade alloys such as G.32 and G.42 which were more highly alloyed than G.18B and possessed greater resistance to scaling together with a considerably improved creep strength. They can withstand operating temperatures more than 100° C higher than can be used with G.18B.

The general practice in this country at the present time with regard to nozzle guide vanes is to use precision cast material. Again in this connection Jessops have done a considerable amount of development work and among their most successful nozzle guide vane materials are G.19 and G.34, which is a cast version of G.32 and also G.39. The last alloy is designed specifically for very high operating temperatures, has adequate resistance to scaling at temperatures up to 1,100° C and, furthermore, is particularly resistant to thermal shock.

Up to the present time, however, the requirements of blade materials for aircraft turbines have, in the main, been met by a series of alloys developed by the Mond Nickel Company. These alloys are based primarily on the 80 per cent nickel and 20 per cent chromium solid solution

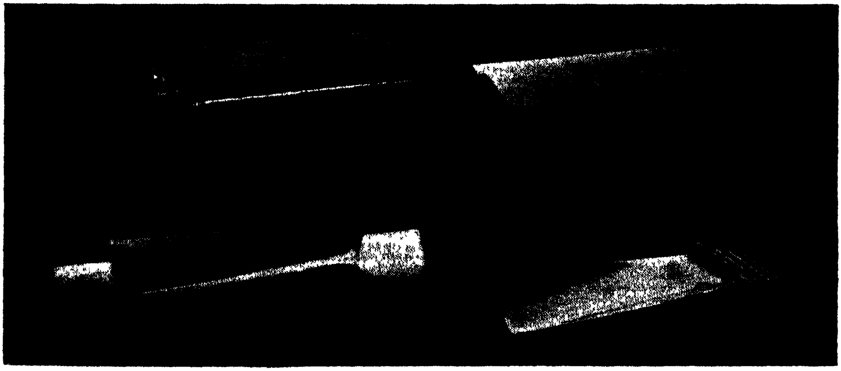
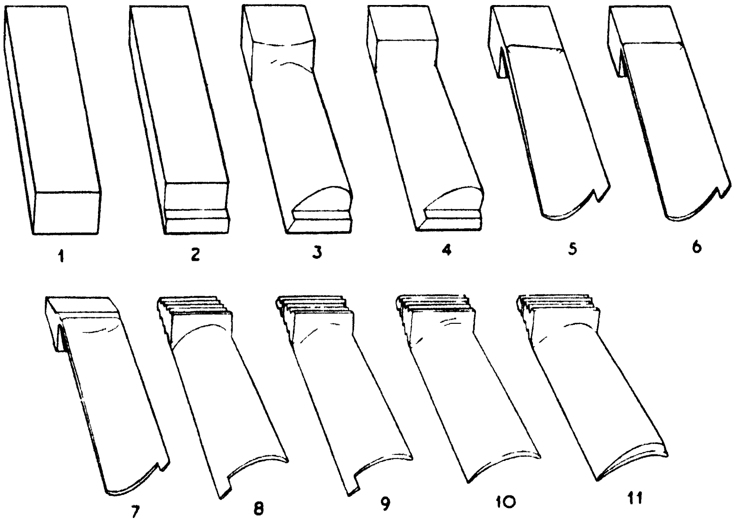


Fig. 8 : 8—A Goblin turbine nozzle vane (top) and a turbine blade shown 'as stamped' and 'as finished.' Both are of Jessop's G. 18 B and produced by the B.S.A Co. Ltd.

with, in the case of the lastest alloys, some 20 per cent of the nickel replaced by cobalt Since 1908 such alloys have been used in continuous or intermittent service at high temperatures in the electrical heating industry, and for the present application the composition was modified in a way which retained the resistance to corrosion and oxidation shown by the basic compositions, but increased their resistance to creep deformation so that they could be used in the aircraft gas turbine.

Nimonic 75 was produced for the early Whittle jet turbines and was intended for an operating temperature of 750° C. It owed its creep resistance to the effect of heat treatment on the solution and precipitation of carbides of titanium. At that time, however, operating temperatures of turbine blades were of the order of 650° C, at which temperature Nimonic 75 did not show as favourably as at 750° C. Physical properties differ somewhat from those of the usual austenitic heat-resisting steels. It has a specific gravity of 8·35 as compared with 7·9–8 0 for steel and the thermal expansion coefficient is lower. It can be readily fabricated to sheet or tube and can be rolled to blade section.



1. Cut slug to length 2 Mill location step 3 Grind convex form· 4. Square platform face (convex side). 5. Mill concave form 6. Square platform face (concave side) 7 Rough mill root-form 8. Grind fir-tree root 9. Grind blade to width, form peening lugs 10. Grind to length: 11. Grind tip-relief.

Fig. 8 : 9—Sequence of machining operations on Bristol Proteus turbine blades in Nimonic 80A.

Research conducted during the subsequent development period showed that titanium, in the comparative absence of carbon and given an appropriate heat treatment, could improve the high temperature properties of nickel-chromium alloys at 650° C as well as at 750° C.

PHYSICAL PROPERTIES

Alloy	*Nimonic* 75	*Nimonic* 80 *Nimonic* 80A	*Nimonic* 90
Specific gravity	8·35	8·2–8·25	8·27
Melting range, °C	1390–1420	1360–1390	1360–1390
Specific heat, *c.g.s. units*			
20–100°C	0·11	0·103	—
20–900°C	—	0·128	—
Thermal conductivity, *c.g.s. units*			
at 100°C	0·032	0·029	0·030
at 900°C	0·071	0·066	0·069
Mean coefficient of thermal expansion, *millionths per °C*			
20–100°C	12·2	11·9	11·6
20–200°C	13·0	12·7	12·6
20–300°C	13·4	13·0	12·7
20–400°C	13·8	13·5	13·5
20–500°C	14·1	13·7	13·7
20–600°C	14·7	14·0	14·2
20–700°C	15·4	14·5	15·0
20–800°C	15·5	15·1	16·0
20–900°C	16·0	15·8	17·0
Electrical resistivity, *microhms/cm/cm² at 20°C* ..	109	124	115

Aluminium was also of value and much investigation was necessary to determine the optimum titanium and aluminium contents and the appropriate heat-treatment cycles to furnish the best balance of high temperature properties whilst retaining an adequate degree of workability.

The new alloy was designated Nimonic 80, and first tried in 1942. The chart in Fig. 8 : 10 indicates the advance which the alloy showed when compared with other alloys then available. Since then research has been continued, and a series of alloys have been developed, the latest being designated Nimonic 95. The continuous improvement in the pro-

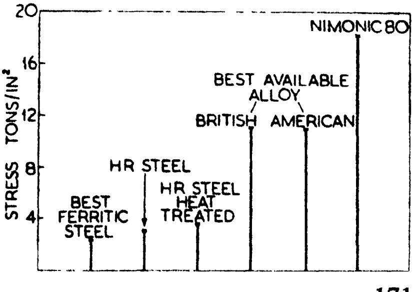


Fig. 8 : 10—Improvement in creep resistance to year 1941. Stress 0·1 per cent in 300 hr at 650°C.

COMPOSITION AND CREEP CRITERIA SPECIFIED FOR THE NIMONIC SERIES OF ALLOYS

	Nimonic 75	*Nimonic F*	*Nimonic* 80			*Nimonic* 80*A*	*Nimonic* 90
	Ministry of Supply Specification						
	D.T.D.703	D.T.D.714	D.T.D.725			D.T.D.736	D.T.D.747
Composition, *per cent*							
Carbon, min ..	0·08	0·08	—			—	—
,, max ..	0·15	0·15	0·1			0·1	0·1
Titanium, min. ..	0·2	0·2	1·8			1·8	1·8
,, max ..	0·6	0·6	2·7			2·7	2·7
Chromium, min ..	18	18	18			18	18
,, max ..	21	21	21			21	21
Aluminium, min ..	—	—	0·5			0·5	0·8
,, max ..	—	—	1·8			1·8	1·8
Silicon, max ..	1·0	1·0	1·0			1·0	1·5
Manganese, max	1·0	1·0	1·0			1·0	1·0
Iron, min	—	5·0	—			—	—
,, max ..	5·0	11·0	5·0			5·0	5·0
Cobalt, min. ..	—	—	—			—	15
,, max ..	—	—	2·0			2·0	21
Copper, max.	0·5	0·5	—			—	—
Nickel	Remainder	Remainder	Remainder			Remainder	Remainder
Creep criteria							
Stress, *tons/in*^{2} ..	—	—	24	19	17	17	19
Temperature $^\circ$C $\pm 2\frac{1}{2}^\circ$ ..	—	—	650	700	750	750	750
Maximum creep rate, *per cent, per hour*	—	—	0·006	0·01	0·01	0·01	0·01
Time to onset of tertiary creep, *hr*	—	—	75	50	25	50	50
Time to rupture, *hr*	—	—	100	75	35	75	75

perties available to the engine designer as each member of the series was announced is indicated by the comparison of the creep properties shown in the accompanying tables. These improvements are the result of a combination of the intensive research by the Mond Nickel Company and the development of suitable working techniques by Henry Wiggin & Company who are responsible for the commercial production of the alloys. Since 1942 one or other of the series has been the standard material for the rotor blades of British aircraft turbines, and indeed for the majority of turbines produced throughout the world.

The Nimonic alloys are essentially forging alloys and are generally produced in the form of bar or drop stampings, but they are also available in the form of sheet or tube.

While machining of this alloy series is not easy, it has been brought to a commercial basis by the use of special techniques. Its merit lies not only in its amenability to manufacturing processes, but also in its resistance to fatigue, which is difficult to obtain in any cast alloy.

The alloys are now produced to meet the requirements of Ministry of Supply D.T.D. specifications covering, primarily, composition and creep properties, and the essential details of the specification are given in the accompanying table. They are, in fact, the only alloys produced to meet a specification laying down requirements covering resistance to creep at high temperatures.

It is unwise to base judgment of a material on its behaviour at a temperature different from that chosen in the design. The most severe conditions of stress and temperature occur during take-off and climb but, since most of the working life is under less arduous cruising conditions, the properties of the material must be considered at both upper and lower orders of stress and temperature. For design purposes, a series of tests is carried out at a variety of stresses and several temperatures covering the working range. The creep stresses determined for given temperatures, times, and stresses are plotted in a family of curves, such as in Fig. 8 : 11, referring to Nimonic 90 at 815° C.

In the two tables overleaf are given the creep properties of the alloy series and data covering the physical short-time high-temperature tensile properties.

Typical creep-extension-time curves for Nimonic 80A at 700° C and 750° C are shown in Fig. 8 : 12, and indicate the creep properties of this particular alloy over long periods of time. As the alloy was not specifically developed for such long-term service those properties can undoubtedly be improved upon by suitable modification of composition or heat-treatment. Further tables give the fatigue properties of the alloy series and torsion tests of Nimonic 80 at high temperatures. The

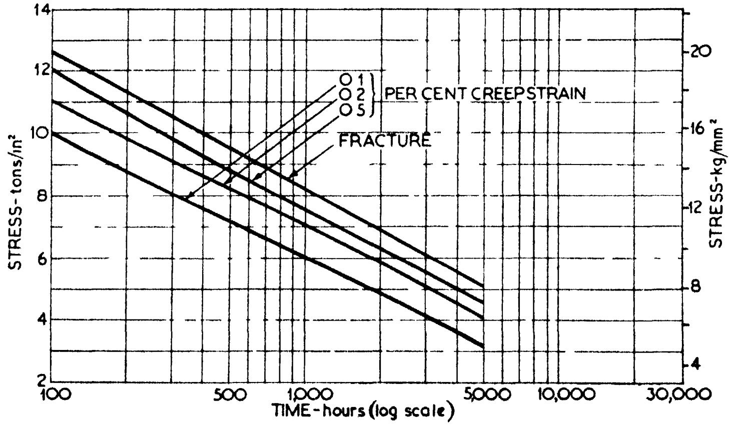


Fig. 8 : 11—Design data for Nimonic 90 at 815°C.

CREEP CHARACTERISTICS OF THE NIMONIC SERIES OF ALLOYS

Alloy	*Temp.* °C	*Stress (tons/in^2) to produce creep extension of*																			
		0·1 *per cent in hours*					0·2 *per cent in hours*					0·5 *per cent in hours*					*Rupture in hours*				
		100	300	1000	5000	10000	100	300	1000	5000	10000	100	300	1000	5000	10000	100	300	1000	5000	10000
Nimonic 80	600	29·5	26·0	21·5	18·5	17·5	32·0	29·5	25·5	21·0	19·5	(33·0)	31·0	28·0	23·5	22·0	(33·5)	32·5	29·5	24·5	23·0
	650	21·0	19·0	16·5	(13·5)	(12·0)	24·5	22·0	19·5	15·5	14·0	26·5	24·0	21·5	(17·0)	(15·0)	28·0	25·5	22·5	18·0	16·0
	700	16·5	14·0	11·0	7·0	5·0	19·0	16·5	13·5	9·0	6·5	20·5	18·0	15·0	10·5	9·0	22·0	19·0	16·0	11·5	9·5
	750	10·0	8·0	6·0	3·5	(2·0)	12·5	9·5	7·0	4·0	(3·0)	14·5	11·5	8·5	5·0	(3·5)	15·0	13·0	9·5	5·5	(4·0)
Nimonic 80A	700	19·0	15·5	11·5	6·0	3·5	23·0	19·5	15·5	10·0	8·0	24·5	21·5	17·5	12·0	9·5	25·5	22·0	18·0	12·5	10·5
	750	14·5	12·0	9·0	4·5	(3·0)	16·5	13·5	10·5	6·5	5·0	17·5	14·5	11·5	7·5	6·0	18·0	15·0	11·5	7·5	6·0
Nimonic 90	650	31·0	29·0	26·0	22·5	—	33·0	30·5	27·5	24·0	—	33·5	31·0	28·0	24·5	—	34·0	31·5	28·5	25·0	—
	700	24·0	21·5	19·0	16·0	—	26·0	23·5	20·5	17·0	—	27·0	24·0	21·0	17·5	—	27·5	24·5	21·5	18·0	—
	750	17·0	14·5	12·0	9·0	—	18·5	16·0	13·5	10·0	—	19·5	17·0	14·5	10·5	—	20·0	17·5	15·0	11·0	—
	815	10·0	8·0	6·0	3·0	—	11·0	9·0	7·0	4·0	—	12·0	10·0	7·5	4·5	—	12·5	10·5	8·0	5·0	—
Nimonic 95	750	20·0	16·5	13·0	9·2	—	22·0	18·0	14·5	10·0	8·0	23·0	19·5	15·6	10·5	—	23·5	19·5	16·0	11·0	—
	815	11·0	8·5	6·0	—	—	12·0	10·0	7·5	—	—	13·0	10·5	8·0	—	—	13·5	11·5	8·5	—	—
	870	7·0	5·0	2·7	—	—	8·0	6·0	4·0	—	—	8·5	7·0	5·0	—	—	9·0	7·0	5·5	—	—
	925	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	5·0	3·8	2·2	—	—

TYPICAL SHORT-TIME HIGH-TEMPERATURE TENSILE PROPERTIES

Alloy	*Property*	*Temperature °C*											
		20	200	300	400	500	600	700	750	800	815	900	1000
Nimonic 80	0·1 per cent proof stress, tons/in^2	—	39·5	39·5	34	36·5	36·5	35	—	26·5	—	—	—
	Maximum stress, tons/in^2	—	66	64	57·5	57·5	48·5	38	—	31·5	—	—	—
	Elongation per cent on 4√Area ..	—	45	44	45	45	25	7	—	8	—	—	—
	Reduction of area, per cent	—	36	41	43	42	19	12	—	10	—	—	—
	Modulus, millions of lb/in^2	—	31·5	30·5	28·5	27·5	26·5	24	—	19·5	—	—	—
Nimonic 90	0·1 per cent proof stress, tons/in^2 ..	45	—	—	—	—	40	37	32	28	26	—	—
	Maximum stress, tons/in^2	74	—	—	—	—	59	52	46	38	34	—	—
	Elongation, per cent on 4√Area ..	39	—	—	—	—	26	20	15	7	8	—	—
	Reduction of area, per cent	20	—	—	—	—	21	7	5	4	4	—	—
	Modulus, millions of lb/in^2	31	—	—	—	—	26	25	24	22	21	—	—
Nimonic 95	0·1 per cent proof stress, tons/in^2 ..	51·6	48·5	46·5	46·3	46	44·3	43	—	31·1	—	17·1	4·12
	Maximum stress, tons/in^2	81·9	77·9	75·1	72	72·1	64·7	51·5	—	40·1	—	21·9	6·2
	Elongation, per cent on 4√Area ..	25·1	20·6	20·6	21·4	23·6	13·7	4·5	—	4·5	—	10·9	59·3
	Reduction of area, per cent	24·1	21·9	24·6	25·3	28·1	19·5	7·1	—	6·4	—	12·5	48
	Modulus, millions of lb/in^2	30·5	30·6	29·3	29·3	27·3	26·4	25·1	—	22	—	17·9	14·1

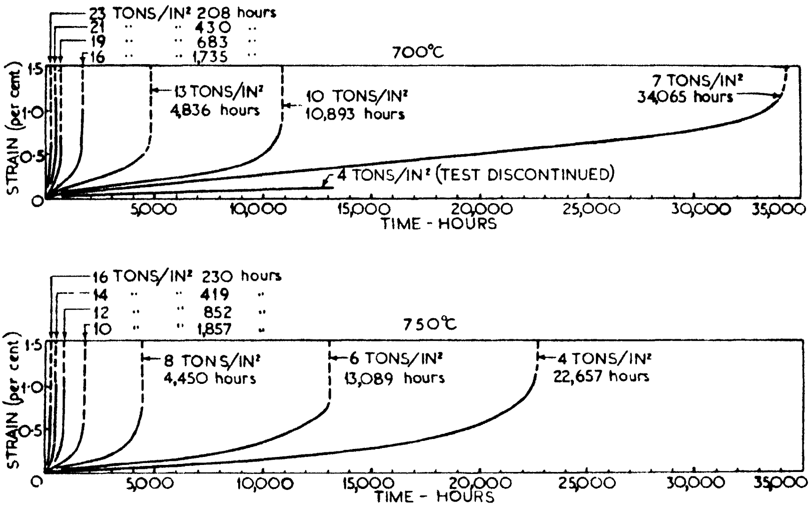


Fig. 8 : 12—Long-time test data for Nimonic 80A at 700° C (top) and 750° C.

improvement in the properties available to the engine designer as the alloy series was developed is apparent.

Nimonic 95 was produced to meet the demands of turbine designers for a blade material to withstand still higher gas temperatures. The alloy is similar to Nimonic 90 but its creep-resistance at elevated temperatures

FATIGUE PROPERTIES OF THE NIMONIC SERIES OF ALLOYS

Tensile-Compressive or Fluctuating Tensile Loading

Alloy	*Temperature* °C	*Stress Condition, tons/in²*		*Endurance in Cycles*
Nimonic 80	20	0 ± 22·0	—	Approximately 40 × 10^6 stress cycles in 300 hr
	600	0 ± 19·0	20·0 ± 13·5	
	650	0 ± 18·3	14·0 ± 9·0	
		—	16·0 ± 8·0	
	700	0 ± 16·7	12·0 ± 8·0	
	750	0 ± 12·4	9·0 ± 5·0	
	800	0 ± 9·0	—	
Nimonic 80A	700	0 ± 18·1	13·4 ± 13·4	Approximately 36 × 10^6 stress cycles in 300 hr
		—	20·4 ± 10·2	
	750	0 ± 14·8	9·8 ± 9·8	
		—	12·8 ± 6·4	
	800	0 ± 12·5	—	
Nimonic 90	700	0 ± 19·2	—	Approximately 36 × 10^6 stress cycles in 300 hr
	750	0 ± 16·7	—	
	815	0 ± 12·2	—	

RESULTS OF TORSION TESTS ON NIMONIC 80

Temperature °*C*	*Limit of Proportionality, tons/in*^{2}	*Proof Stress, tons/in*^{2}			*Modulus of Rupture, tons/in*^{2}	*Angle of Twist to Fracture, degrees*	*Modulus of Rigidity, millions of lb/in*^{2}
		0·1 *per cent*	0·2 *per cent*	0·5 *per cent*			
20	20·2	26·8	28·2	30·3	64·8	1055	12·00
200	19·0	25·5	26·7	28·4	62·3	1128	11·30
250	17·2	25·0	26·5	27·8	60·6	1105	11·30
350	17·4	25·0	26·4	27·7	60·6	1335	11·55
450	17·2	24·5	25·9	27·5	57·2	1100	11·20
550	19·8	26·9	28·1	29·7	57·2	977	10·90
650	21·6	27·6	29·1	30·8	48·8	354	10·40
700	18·2	25·1	27·0	29·4	42·1	248	9·35

Test piece: 0·300 in dia, 3·04 in parallel length, 2 in extensometer gauge length, loaded at 0·84 ton/in^2 surface shear stress per min.

was improved by an increase in the content of the hardening elements. Any raising of the temperature at which an alloy is serviceable necessarily involves a reduction in the temperature interval during which hot-working can be carried out, since additions to the amounts of alloying elements do not produce a significant rise in the melting temperature. Only by meticulous control of composition, melting, casting and production methods is it possible to maintain forgeability adequate for practical purposes.

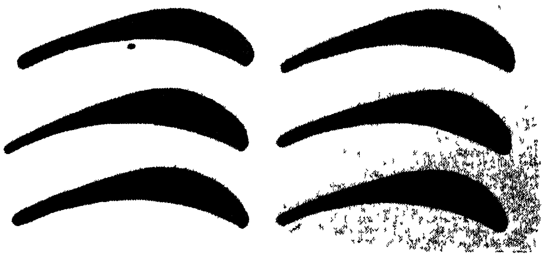


Fig. 8 : 13—Various configurations of cooling passages in extruded blade forms of Nimonic 90 alloy.

As operating gas temperatures are raised in the quest for higher efficiency, a stage may be reached when alloys for more onerous service cannot be forged by normal economic techniques. Accordingly more attention is likely to be directed to systems of blade cooling. In this connection Henry Wiggin & Co. Ltd., the producers of the Nimonic series of alloys, has developed a novel method of forming a hollow, air-cooled blade. The cast ingot is machined to a uniform cylindrical billet and one or more holes are drilled longitudinally. These holes are filled with a suitable material and the ingot is then extruded to airfoil section on the Ugine-Sejournet system, using glass as the die lubricant. Since the holes are both deformed and displaced during extrusion, their individual size and collective location must be carefully arranged and, of course, proved by trial. An envelope about 0·03 in thick is allowed on the section for machining. Subsequently the filler material is removed by immersion in an acid bath that is innocuous to the blade alloy. By breaking down the coarse as-cast structure and thus improving forgeability, the extrusion process extends the composition range of alloys that can be commercially worked. Nozzle and stator blades produced by this method in Nimonic 80A and Nimonic 90 and cooled by an adequate air-flow, show reductions in peak temperatures up to 300° C as compared with normal solid blades.

NOZZLE GUIDE VANES

The guide vanes at the entry to the turbine are exposed to higher gas temperature than the rotor blades but are stressed only relatively lightly to the order of 1·5 tons/in^2. Working temperature under maximum operating conditions may be in the region of 850° C and consequently creep extension can be a problem. The outstanding difficulty, however, arises from the stress gradients caused by rapid heating and cooling encountered by the frequent starts and shut-downs of aircraft turbines. The behaviour of an alloy under these conditions is not revealed by the normal creep tests. In particular, the stator ring supporting the vanes exercises a local cooling effect and produces a steep thermal gradient on starting which may eventually lead to buckling or cracking of the vanes.

Nimonic 75, 80, and 80A alloys in the rolled, forged or cast form are used for the manufacture of nozzle guide vanes, and their high resistance to oxidation makes them suitable for the highest gas temperatures used so far. They can be precision-cast by the ‘ lost-wax ’ process, thus rendering possible the production of relatively intricate guide vane assemblies. Stator rings forged in Nimonic 75 combine useful creep strength with low expansion characteristics and thus tend to minimize distortion and consequent loss of efficiency.

COMBUSTION CHAMBERS

A number of heat-resistant metals and alloys were studied and used experimentally for the combustion chambers of the early Whittle turbojets. Finally Nimonic 75, developed during the Second World War, was adopted for the flame tubes of combustion chambers. Since that time it has been used for combustion components on most British aircraft units and also for a number of the large land and marine turbines in course of development.

Its performance under such exacting conditions is due to a combination of properties. High creep and fatigue strengths at the operating temperature are allied with a low coefficient of expansion. All heat-resisting alloys depend on the formation of an initial oxide film for their resistance to continued oxidation. Chromium and aluminium, two constituents of the alloy, are of importance in the formation of this protective film which, in the case of Nimonic 75, is firmly adherent. The coefficient of expansion, similar to that of mild steel, is of advantage in reducing stress due to thermal gradients and, coupled with high mechanical strength, tends to minimize the risk of distortion. This feature is of some importance as the functioning of a combustion chamber is dependent on the shape of its components, particularly the flame tube.

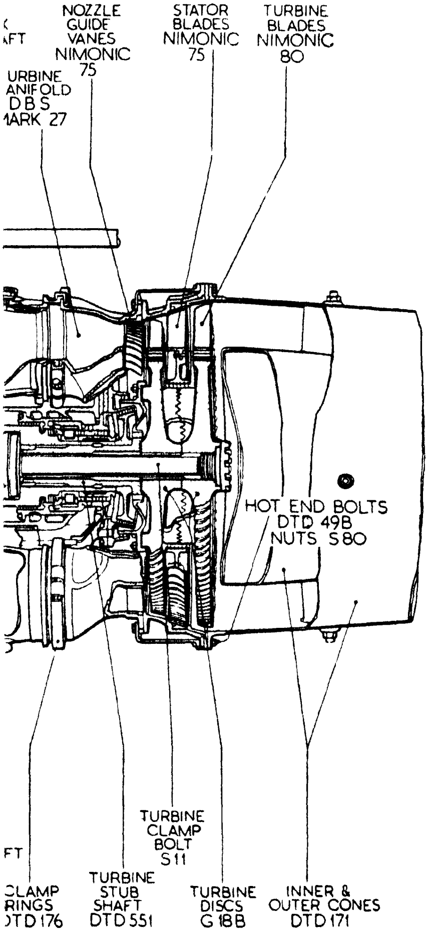
Nimonic 75 is forged and rolled to sheet and can be hand-beaten, spun, pressed or rolled at room temperature. Resistance welding, or argon-arc welding, is commonly used for this material as its high electrical resistivity reduces the current required to effect a weld and its low thermal conductivity localizes the heat. To facilitate welding, the carbon content of sheet material is normally held to a maximum of 0·12 per cent.

In 1953 the producers introduced a sandwich material comprising a sheet of copper rolled between two sheets of Nimonic 75. Designated Nimoply 75, this clad sheet retains the characteristic properties of Nimonic 75 but has a much higher heat conductivity. Flame tubes of this material dissipate the heat of combustion more rapidly and ensure a more even temperature distribution, with consequent less risk of distortion, in the chamber.

Combustion chambers of the annular type, not using flame tubes and generally operating at slightly lower temperatures, have been fabricated from austenitic steel sheet of Immaculate 5. In America, annular combustion chambers have been made of high nickel-chromium alloys of the Inconel type, 18/8 chromium-nickel-molybdenum steels and S.816, a more recently developed alloy.

COMPRESSOR IMPELLERS

Although the compressor is not thermally stressed to the same extent as the turbine, the impellers of centrifugal compressors for turbine units presented some problems. Their production necessitated the manufacture of a new type of aluminium-alloy forging. The material had not to be subject to size effect nor to reduced properties transverse to the grain flow, and at the same time possess good hot strength and creep resistance at temperatures up to 250° C. The impellers required were from 22–33 in diameter and from 10–14 in thick and this implied solid forgings weighing from 400–1,000 lb each possessing the mechanical properties hitherto expected from a one inch square test bar. Rotors for axial flow compressors necessitated hollow forgings up to 15 in diameter and 20–30 in long with a wall thickness of about 3 in.



Mo	*V*	*W*
0 65*	0 25*	1·0*
—	—	—
—	—	—
0·5*	0·25*	1·0*
—	—	—
—	—	2·0–4·0
Opt.	Opt.	Opt.
Opt.	Opt.	Opt.
·4–0·7	0·25*	—

The Rolls-Royce alloys were considered most suitable due to their good constituent dispersion and to their adaptability to the chilling process without undue skin segregation. R.R.56, R.R.58 and R.R.59 have been used. The nominal percentage composition of R.R.59 is : copper 2·2, magnesium 1·5, nickel 1·2, iron 1·0, silicon 0·9 and titanium 0·05.

Later study focused attention on the L42 low silicon alloy. Difficulties had previously been experienced in casting or working this material but special high purity aluminium was employed and a new technique of forging was developed and the manufacturing difficulties were rapidly overcome. A typical percentage composition of this alloy is : copper 2·2, magnesium 1·6, nickel 1·2, iron 1·0, silicon 0·1 to 0·2 and titanium 0·1.

High Duty Alloys, Ltd. proved that impeller failures were mainly due to heavy stresses set up during heat treatment and recommended the provision of a central hole in the forging to reduce the differential cooling rate on quenching. While this method results in increased centrifugal stressing in operation, it has completely eliminated failures in service.

Axial-flow compressors may have both drums and blading of light alloy as the temperatures are relatively low. As operating conditions are stepped up, however, the trend is towards the use of aluminium bronze, manganese-bronze or chromium-nickel steels, notwithstanding the added weight. The necessity to operate in salt-laden atmospheres puts a premium on resistance to corrosion and, in the case of large land and marine turbines, increased recuperation leads to higher working temperatures. On aircraft units there always exists the danger of foreign bodies being drawn into the compressor. In this respect the blading of an axial flow compressor is less robust than the impeller of a radial flow unit and an addition of mechanical strength is of undoubted advantage. The trend is towards the wider use of steel for blades, rotors, and even casings, particularly at the high-pressure end of a compressor.

SELECTION OF MATERIALS

The operational efficiency of a turbine unit depends as much on the wise choice of materials for the various components as on the careful attention to design. It may be suggested that interdependence of materials and design is probably of greater importance in the turbine than in any other type of power unit. The careful selection of materials is well shown in Fig. 8 : 13, in which the different steels and some of the non-ferrous materials used for component parts of the Armstrong Siddeley Mamba turboprop unit are labelled. The compressor is of the combined drum and disc type with the blades riveted between the rims of pairs of S.11 steel discs shrunk on to the inner drum which is also of steel. Both front and rear shafts, bolted to the drum, are of S.11 steel and both rotor and stator blades are of R.R.57 light alloy. With the aim of reducing weight, rotor drums of duralumin are being investigated.

In the combustion chambers the flame tubes and the primary air tubes of Nimonic 75 are housed in cans of D.T.D.171. The nozzle manifold connecting the ends of the combustion chambers to the turbine entry is cast in D.B.S. 27 steel.

The turbine has two separate discs of G.18B, with a tubular shaft stub of D.T.D.551 bolted to the forward face of the first stage disc, drawn up to the main shaft by a substantial clamp bolt of S.11. Rotor blades are of Nimonic 80 and Nimonic 75 is used for the nozzle guide vanes or stator blades.

From the front of the compressor the airscrew is driven through an epicyclic reduction gear. In this unit gears and pinions are all of S.82 steel while the torsion shaft, the satellite carrier and the airscrew shaft are in S.11 steel. At various points in the unit are labyrinth seals, notably at the front and rear of the compressor and at the front and interstage position of the turbine. Pure nickel strip is used for seals at low temperature locations but at the hot end of the unit F.D.P. stainless steel strip is fitted as it is more resistant to corrosion by the sulphur present in the combustion products.

Selection of materials, and even the design of components, is unavoidably influenced by the availability of specific materials and, particularly, of relatively scarce alloying elements. A vivid example of this was provided by the efforts of German engineers to evolve blade designs that were economical as regards the use of such scarce materials (see Chapter 5).

With the widespread adoption of turbine power units for both civil and military aircraft and the current extensive rearmament programmes throughout the countries of the world, the subject is still of prime importance. Conservative and efficient utilization of materials in general were studied by a specialist productivity team which visited the U.S.A. to enquire into and compare methods. In their subsequent report, ' The Saving of Scarce Materials,' the team pointed out that a higher proportion of scarce ' strategic ' materials was used in American jet units than in the British counterparts.

It appears that in the U.S.A. there have been rival schools of thought. One advocates the design of units for a short, war-time expectation of life in order to conserve materials. Against this viewpoint the other school suggests that the airframe, in numerous instances, will outlast the power units and consequently needs replacement units. Thus the material economy would be illusory and manufacturing facilities would be overloaded. Accordingly, it favours the British practice of building for a more reasonable expectation of working life.

CHAPTER 9

Research, Testing and Maintenance

WHILE less common now than in the early years, the impression that the aircraft gas turbine is a remarkably simple prime mover still persists. Structurally and functionally it is certainly less complicated than the high-powered reciprocating engine that was its immediate predecessor, but its design calls for the most meticulous and exhaustive study. Merely to equal or slightly exceed the performance of the reciprocating engine would require only a very simple unit. The tremendous increase in power output, however, together with high efficiency and long life has made possible flight at hitherto unattainable altitudes and velocities and introduced new and difficult problems.

To some extent the illusion of simplicity has tended to continue as a result of the exceptionally rapid progress of the gas turbine. In the short span of about 12 years it has displaced the long-established piston engine for all but minor military aircraft and already British turbine-powered air liners are in commercial operation on world routes. This has only been achieved by an unprecedented concentration of research, resources and finance—truly a national effort sustained with unabated war-time intensity.

Of gas turbines more, possibly, than of any other form of mechanical design, it may be said that if the mathematics are not fundamentally correct at the start, then no amount of effort can make the end product a success. This is accentuated by the fact that the mathematics are extremely complex and an enormous number of mathematician man-hours must be expended to achieve seemingly trifling progress. For this reason resort is had to electronic calculating equipment, such as the A.C.E. automatic computing engine developed by English Electric and the N.P.L., and the numerous commercial computors now available. Fed with appropriate data by a skilled operator, such apparatus can solve in a fraction of a minute a complex problem that would necessitate a full week's work by an accomplished mathematician.

References to supersonic velocities, that is speeds in excess of 760 m.p.h. or 1,120 ft/sec at sea level, usually relate to missiles or aircraft but it should also be borne in mind that such velocities are attained by the gas flows in turbines, ramjets and rocket motors. Necessarily the forms and contours of those power unit components subjected to flows of such velocities must be investigated and tested in high-speed wind tunnels.

Continuous-flow tunnels are enormous aggregates, costly to build, operate and maintain, and can be sponsored by only the largest organizations or institutions. An outstanding example is the 6 ft × 6 ft N.A.C.A. supersonic tunnel at the Ames Aeronautical Laboratory at Moffet Field, California. It is capable of producing in the test section conditions equivalent to a velocity of Mach 1 6. The tunnel airflow is generated by a gigantic 8-stage axial compressor driven by two 25,000 h.p. electric motors.

Intermittent-flow tunnels occupy less space and are less costly to build and run but have obvious limitations in operation. One of the more interesting intermittent tunnels is the vacuum-operated type installed at the aerophysics laboratory of North American Aviation Inc., Los Angeles, California. This is based on the design of the Kochel tunnel which was removed from Germany and re-erected at the Naval Ordnance Laboratory at Silver Springs, Maryland. It was with the Kochel tunnel that much of the initial development work on the A-4 long-range rocket was carried out.

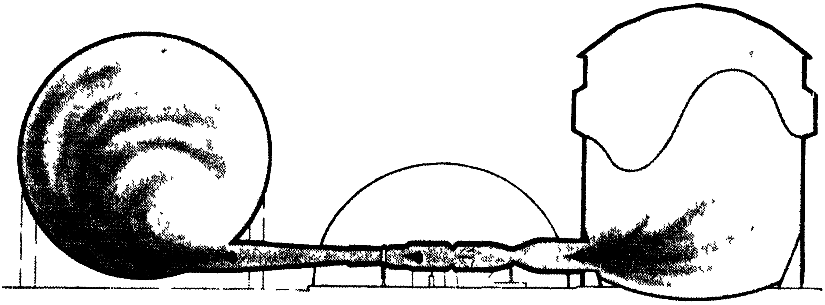


Fig. 9 : 1 Diagrammatic section of vacuum-operated, intermittent-flow, supersonic tunnel.

The North American tunnel, which has a 16 in × 16 in working section, occupies an area 170 ft × 80 ft and the height of the tallest component is 48 ft. This is the storage tank in which 22,500 ft^3 of dry air is accommodated between rubber-impregnated nylon membranes. The air, at a pressure of 0 0046 lb/in^2 in order to raise the upper membrane, is displaced atmospherically through the test section to a 36,000 ft^3 vacuum chamber.

In the test section air velocities of from Mach 1·22 to Mach 5·25, corresponding to sea-level speeds of 930 to 4,000 m.p.h., can be produced for runs of 15 to 18 sec duration. The operating speed is determined by the profiles of the interchangeable nozzle blocks mounted immediately upstream of the model.

Whilst it requires 37·5 min for the two vacuum sets, each comprising

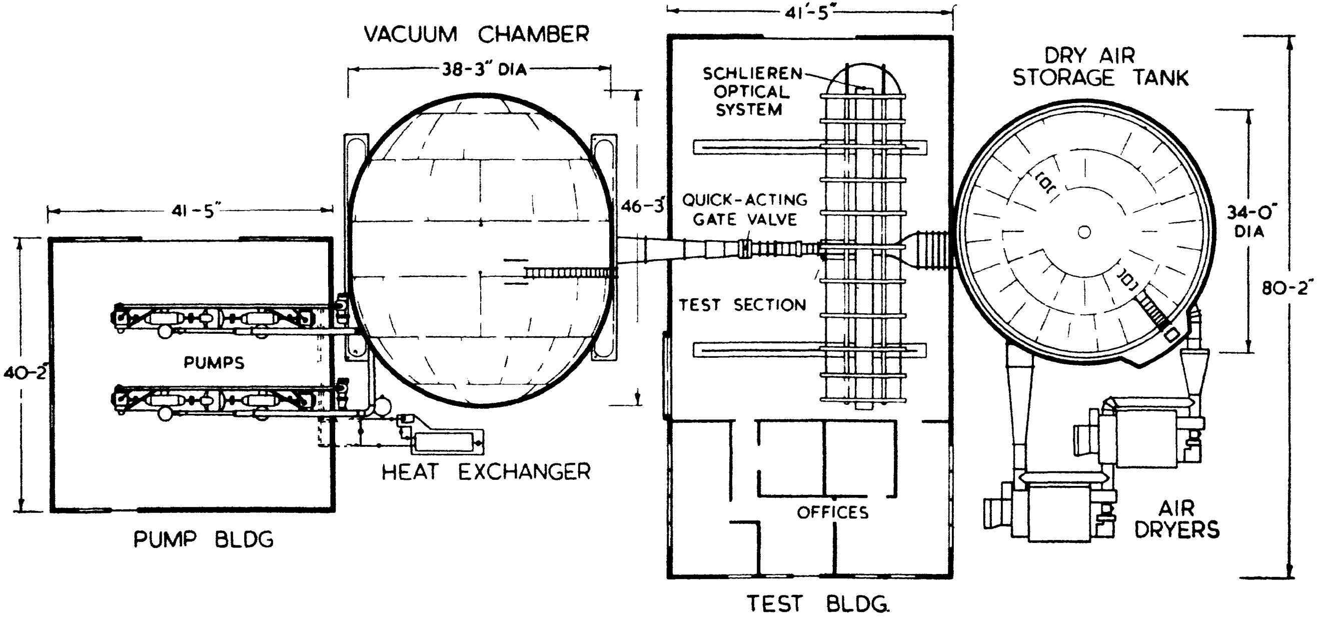


Fig. 9 : 2—Layout of North American intermittent-flow, supersonic tunnel.

a pair of 2-stage pumps driven by a 200 h.p. motor, to reduce pressure by 29·7 in. in a 30·0 in mercury barometer, pumping down between runs is considerably shorter as a partial vacuum remains after each run. With 15 sec runs, according to the operating speed selected, the pump-down time varies from 6·5 to 15·5 min.

High-speed tunnels are usually equipped with Schlieren shadowgraph apparatus for deriving visual and photographic evidence of flow phenomena. A typical installation, at the North American tunnel, consists of a 30 in diameter tube suspended from a steelwork bridge and straddling the test section. The complete bridge structure is mounted on rails running parallel to the test tunnel to permit it to be moved into and out of position. At each end of the tube is mounted a 20 in diameter parabolic mirror. Light from a source adjustably mounted on the tube is passed through a knife-edge slit to the first mirror which reflects parallel rays through the working section to the second mirror. In passing through the test section the light rays are deflected proportionately to the variations in density of the air stream in the tunnel. The second mirror focuses the light beam on a second knife-edge slit so that the airflow patterns are made visible on a screen or recorded on a photographic plate. To enable several observers to view the pattern simultaneously a projection system incorporated with the camera can throw the picture on a large screen.

The characteristic Schlieren picture, Fig. 9 : 4, shows the patterns of shock wave formations about the nose of a small-scale model ramjet intake at Mach 2·1. This photograph was taken at the supersonic tunnel in the Wright Aeronautical Research Laboratory.

Nitrogen has the property of glowing after an electrical discharge is passed through it and at the Langley Aeronautical Laboratory a method of introducing nitrogen through an electrical screen to the air flow in a tunnel has been developed. Thus a glow is produced around the model undergoing test and the shock wave pattern revealed by the Schlieren optical system is heightened.

For tests of such items as ramjets, rockets and guided missiles, extremely high flow velocities are necessary and for this purpose so-called hypersonic tunnels have been developed in the U.S.A. First operated in 1947, the 11 in tunnel at the Langley Aeronautical Laboratory is of the intermittent-flow type described earlier but the air in the supply tank is raised to a pressure of 50 atm (735 lb/in^2). The duration of run is from 60 to 90 sec and reports have been issued of investigations at a Mach number of 6·9.

Since the expansion ratio at the nozzle is so high liquefaction of the air would occur if the supply were at atmospheric temperature. Accordingly, heating equipment is provided to raise the stagnation air temperature to approximately 700° C. At the low air density in the working

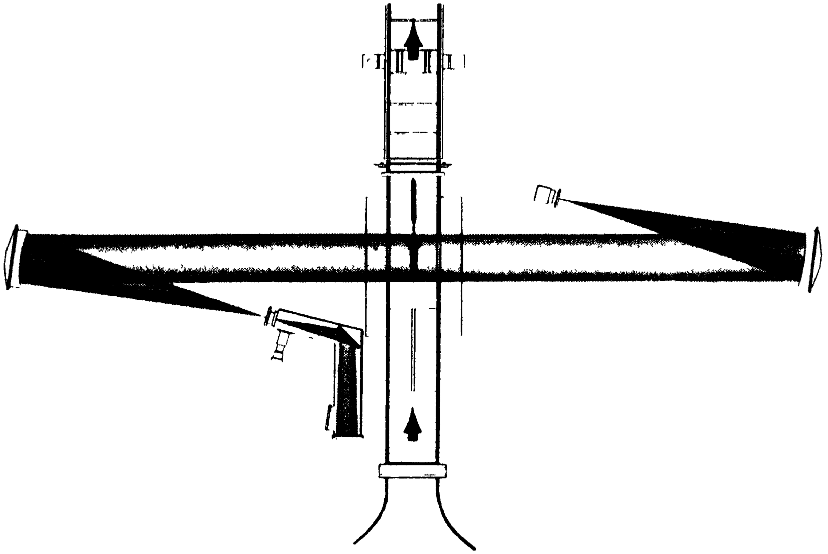


Fig. 9 : 3- Diagrammatic arrangement of Schlieren apparatus. A beam of light is directed on to a parabolic mirror (right) which reflects parallel rays through the test section. Second mirror (left) focuses rays on a knife-edge slit in the camera (centre).

Fig. 9 : 4—Schlieren photograph of shock wave patterns on ramjet model at Mach 2·1.

section—it may fall to 1 mm Hg—the nitrogen glow technique is essential to secure flow visualization by the Schlieren system.

Another tunnel of this type at the Ames Aeronautical Laboratory has a working section of 10 in × 14 in and operates with a Mach range of from 3·0 to 8·0. At the California Institute of Technology a 5 in × 5 in tunnel has been operated at flow velocities exceeding Mach 10·0. In this plant the air supply is raised to pressures as high as 1,000 lb/in^2 by a group of fourteen compressors driven by six electric motors having a total output of 775 h.p.

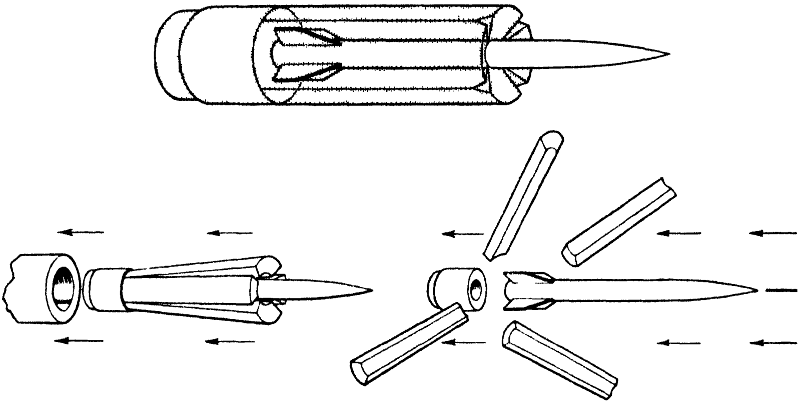


Fig. 9 : 5—Sketch showing assembly of free-flight model in sabots which strip off on leaving the gun barrel.

Also at the Ames Laboratory is a novel free-flight tunnel which offers several advantages for high-speed investigations. The model is fired from a gun located in the diffuser section and directed upstream towards the nozzle. The supply air need not be so highly compressed, for with a flow at Mach 2 in the working section and the model projected at, say, 5,000 ft/sec velocity, the relative Mach number will be about 8. For insertion in the gun barrel the model is assembled in a series of segments of a hollow cylinder, termed ‘sabots.’ Leaving the nozzle these are automatically stripped off and fall away and the model continues in free flight, passing through the tunnel nozzle and being caught in a special trap in the reservoir. The working section is 1 ft × 2 ft × 8 ft long and the flight of the model, in distance and time, is recorded on photographic film by an elaborate optical system.

A subsonic, continuous-flow tunnel that is relatively economical to build and operate has been developed by the English Electric Co. Atmospheric air is drawn through the working section by a standard turbojet and discharged to atmosphere. About one-third of the tunnel air is taken into the power unit whilst the rest is induced by the ejector action of the jet. Part of the hot efflux is fed into the incoming air to lower humidity and thus obviate condensation. The initial plant, powered by a R-R Nene turbojet and producing a flow rate of Mach 0·9, occupied a space only 60 ft × 34 ft × 18 ft high and cost merely a few thousand pounds to build.

At the Handley Page research station a different arrangement is employed for a ‘high sub-sonic’ open-circuit tunnel. The three R-R Nene turbojets supplying the power are housed externally to the tunnel to permit maintenance or inspection whilst working and also to avoid damage in the event of a model being carried away. Air for combustion is obtained directly from the atmosphere and independently of the tunnel flow. Condensation is prevented by recirculating some of the hot exhaust flow, which is introduced through hollow airfoil-section splitter vanes at the intake. In the 4 ft × 3 ft working section the induced air can be accelerated to sonic velocity.

STATIC TESTING

All power units undergo a comprehensive testing and inspection before they are passed for service. As an example may be quoted the test schedule for the widely used de Havilland Goblin turbojet.

(1) Preliminary adjustments and starting test.
(2) Non-stop run of 2 hr, consisting of 5 min at idling speed, 1 hr 45 min at maximum cruising speed, 5 min at climbing speed, and 5 min under take-off conditions.
(3) Acceleration tests from idling speed to maximum r.p.m.

On completion of the preliminary tests the engine is stripped down, inspected and rebuilt. It then goes through a final test consisting of:

(1) A run of 15 min at maximum cruising speed.
(2) Accelerations from idling speed to take-off r.p.m.
(3) Complete thrust calibration over the whole speed range.

Each test house is separate and self-contained and comprises basically the engine compartment, the control room and the exhaust tunnel, while in addition there are pump, filter, rectifier, tank and fan rooms. A turbine on its stand is wheeled in through a door in the near end of the exhaust tunnel, positioned on the floating test platform which is supported on rollers, and secured. A flexible, forked air duct is attached to the bifurcated intake, pipe lines and controls are connected up and exhause detuner duct and tail pipe guard are brought into position The time taken for Goblin units is about 2 hr and removal takes 1½ hr.

Thrust is measured directly through the reaction of the unit or mounting against a Statimeter transmitter button. The movement is minute—about 0·001 in—and the button on production beds is connected by

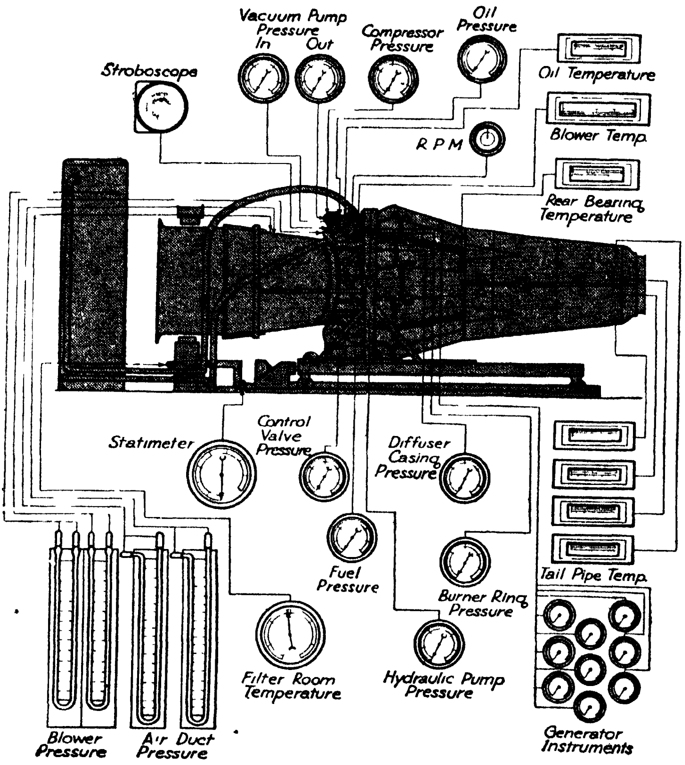


Fig. 9 : 6—Schematic arrangement of connections for instrumentation of de Havilland Goblin test.

capillary tubing to a Bourdon gauge registering in pounds thrust. On the most modern beds, however, a simple dead-weight, direct-calibrating arrangement is used for thrust measurement. The power unit is mounted on a floating platform connected by a system of levers to a large Vandome and Hart weighing machine from which a direct thrust reading in pounds is obtained.

When the unit is running, temperatures, pressures and consumptions are observed and recorded in a sound-proofed control-room in the same way as for any other type of engine. The particular system of stroboscopic engine speed measurement on some beds is interesting and a little out of the ordinary. The set-up is the work of the de Havilland Company, and incorporated are a master tuning fork with electrically-controlled frequency and a three-aperture scanning disc, both in the engine compartment. Rotational speed is indicated electrically in the control cabin and the scanning disc is observed through the double window to the engine compartment. When testing any power unit, the requirement is usually to run at predetermined engine speeds as opposed, for example, to measuring r p m. at given throttle settings or combustion temperatures. The stroboscopic scanning disc allows accurate readings to be taken through its three apertures for thousands, hundreds and tens of r.p m.

Temperatures observed include those of the air-intake duct, compressor delivery, combustion chambers, jet pipe (at four points), rear and front bearings, oil sump, Heywood compressor and, for instrument correction, the control cabin and filter room temperatures. Among the pressures measured are those for diffuser, turbine, intake duct, impeller tip, tail pipe and vacuum pump. The main hydraulic pump is checked in conjunction with a loading valve up to 3,000 lb/in^2, and the Marshall blower for pressure cockpit or cabin has a large receiving tank in the filter room.

With the exception of the exhaust tunnel the test houses are very compact. Intake air is conveniently drawn in through a large duct over the test cubicle and passes through ‘ splitter ’ panels before entering the filter room. In that the engines must often inhale non-filtered air when installed in an aircraft, they are expected to do so on test. The name

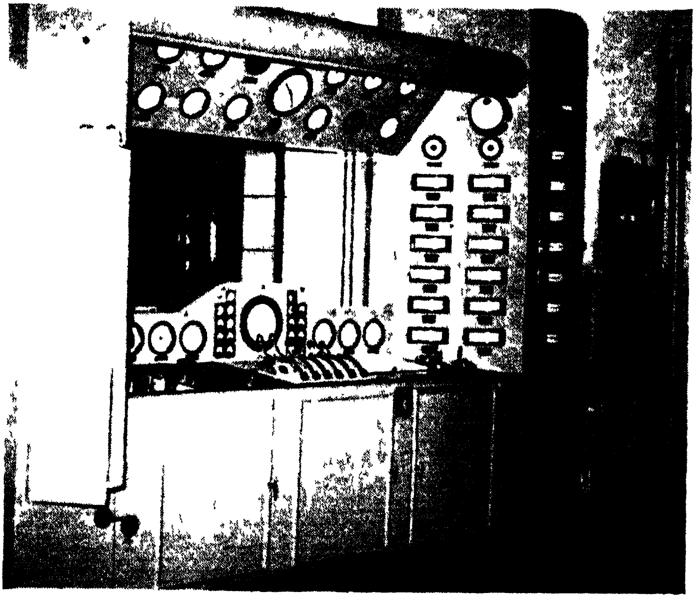


Fig. 9 : 7—Typical Heenan and Froude control desk for turbine unit test house.

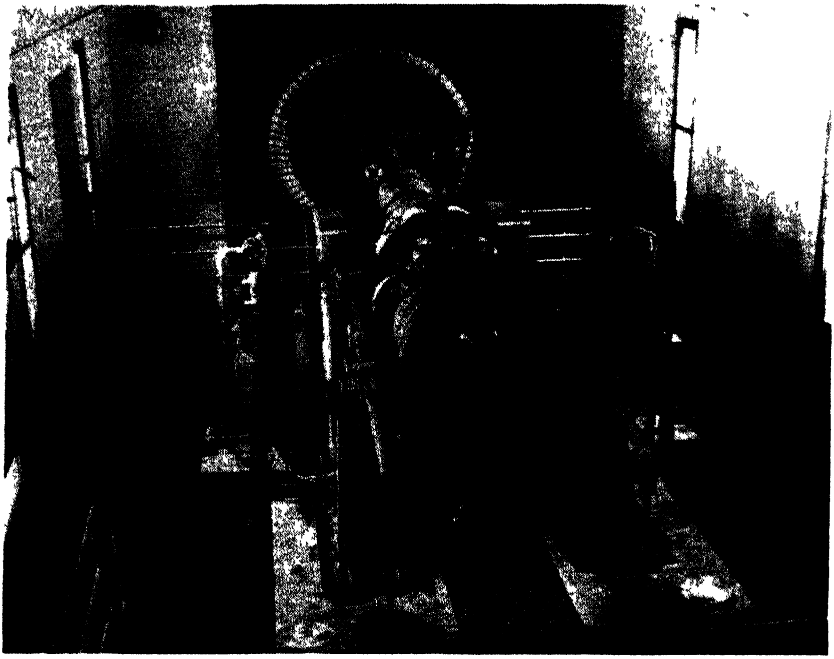


Fig. 9 : 8—R-R Avon on test stand at the Standard Motor Co. Ltd., Coventry.

filter room is, therefore, something of a misnomer for what is in effect a large intermediate compartment in the intake-air duct system. At the end of the exhaust tunnel a brick blast wall is provided to deflect the jet vertically upwards

For the R-R Avon turbojet, produced by several manufacturers as part of the national supply programme, virtually standardized test equipment has been built by Heenan and Froude. Buildings, layout and detail arrangements differ to meet site conditions but the main test beds are of common design and can accommodate units larger than the Avon. They are, of course, equally suitable for the turbojets of other manufacturers.

The unit is mounted in a cradle suspended from heavy cast-iron pillars by steel straps arranged to permit axial freedom of movement while resisting lateral deflection. To meet the contingency of slight deviation of the thrust axis of the power unit from the longitudinal axis of the test stand, the thrust is taken from lateral connections on the cradle to a pair of levers coupled by a stiff torsion shaft. By a leverage system the thrust developed by the power unit is reduced in the ratio of 75 : 1 before transmission by means of a high-tensile steel wire to the Avery pendulum-type weighing machine located in the control room. The mechanism can be checked for accuracy of indication by a direct application of weights.

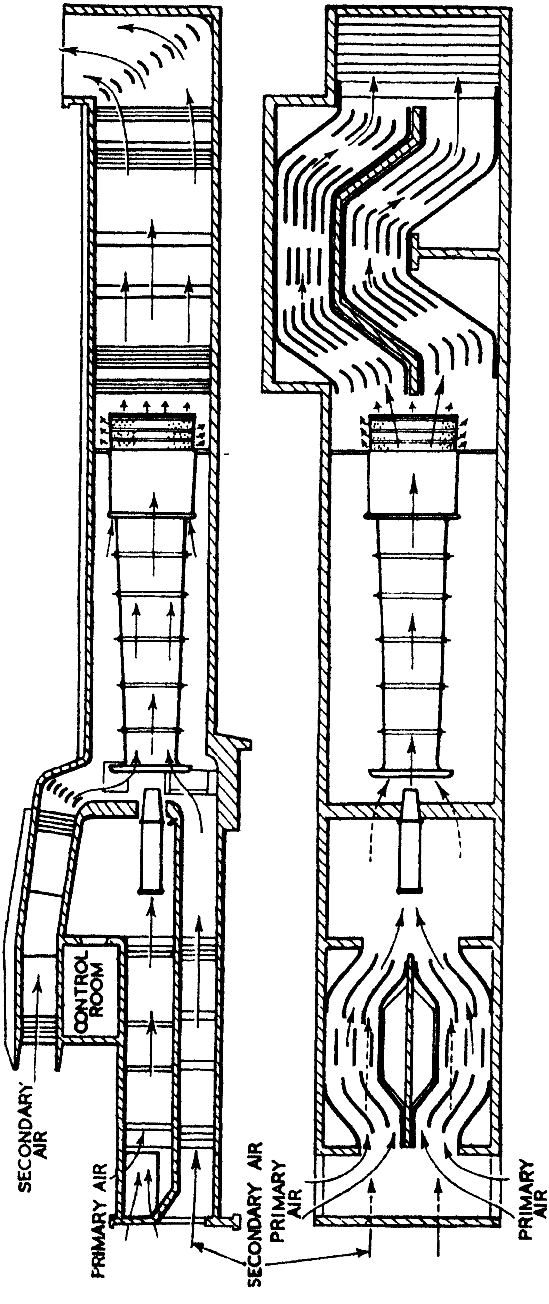


Fig. 9 : 9—Sectional elevation and half plan of Bristol test house.

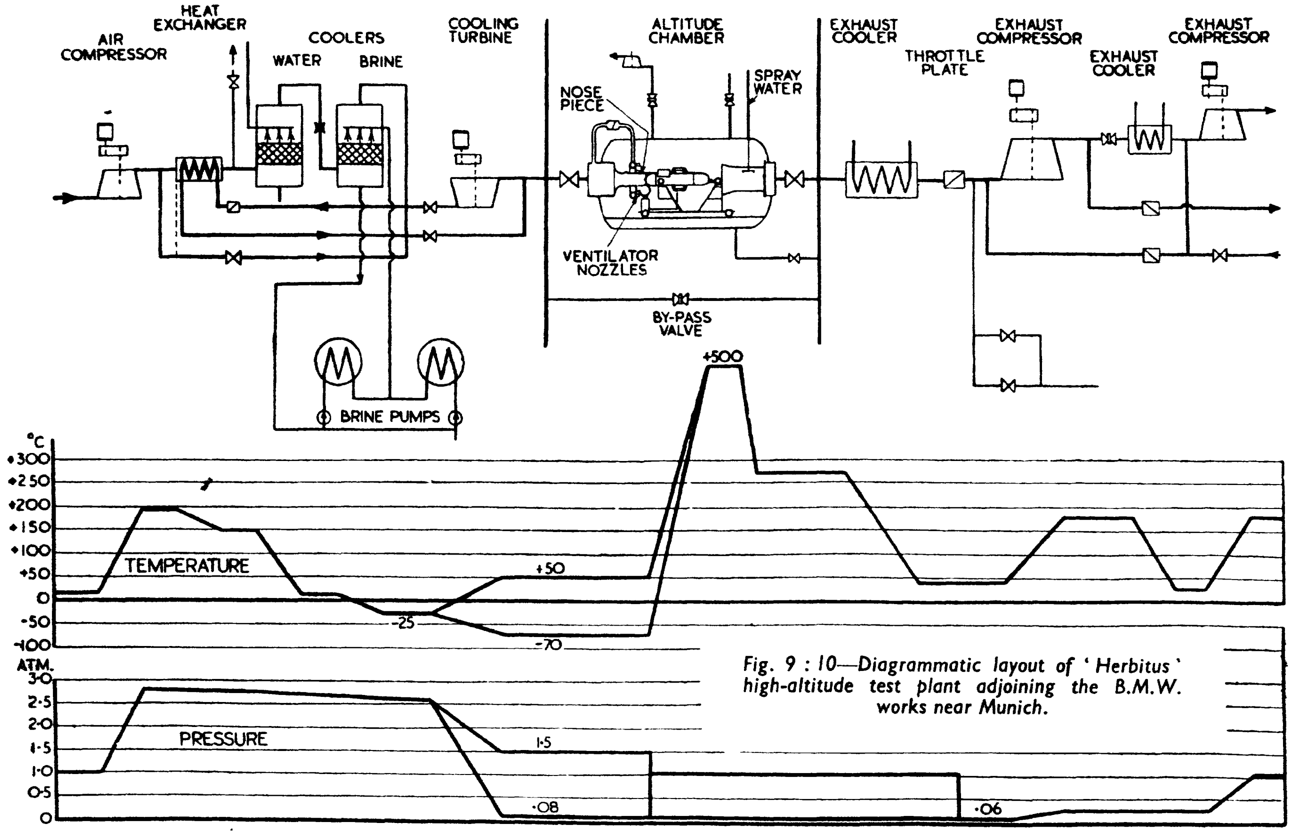


Fig. 9 : 10—*Diagrammatic layout of 'Herbitus' high-altitude test plant adjoining the B.M.W. works near Munich.*

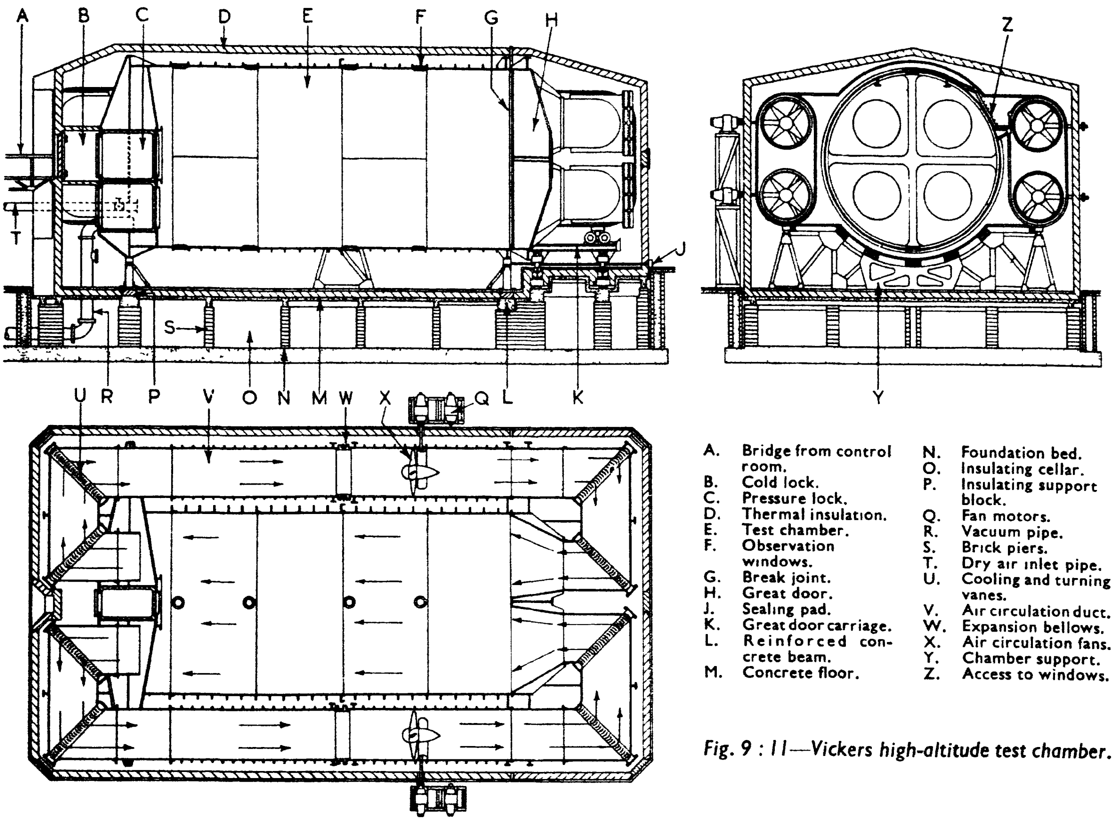
It was stipulated that any error in reading should not exceed 0·1 per cent at maximum load. Check tests on installed plants have shown an average of inaccuracies of less than 0·025 per cent.

The plant comprises main fuel tanks, each having a capacity of 12,000 gal, sunk in a pit remote from the main buildings ; a fuel supply cubicle containing direct-reading flowmeters for main and afterburning supplies ; pump house ; battery, test, and control rooms. Fumes are extracted from all rooms, special fans with flameproof motors being used for the fuel cubicle. A comprehensive CO_2 fire-fighting system is installed, with interlocked change-over controls to shut off the fuel supply, open the CO_2 supply, switch off the ventilating fans and operate an alarm.

Full-power testing of large turbojets creates major problems of noise over a wide range of frequencies. It arises from the enormous air mass flows associated with these units and is generated at the intake as well as at the exhaust. Its reduction to reasonable values not inimical to the amenities of the surrounding neighbourhood calls for elaborate silencing arrangements involving considerable expenditure. Silencing of an engine test station is considerably simpler than silencing an engine mounted in an airframe. The various solutions all cool and decelerate the jet, and break up the noise spectrum into distinct frequency bands which are damped in series. The most-used equipment are Cullum Detuners or acoustic tunnels made by Cementation, Ltd. These units all employ air cooling; water cooling is used in some other countries.

The constructional layout adopted at the Standard R-R Avon test plant follows that developed by Bristol, in conjunction with Cementation Ltd., for beds to accommodate the Olympus turbojet. Air for the power unit is divided and diverted through two cranked passages furnished with vertical cascade vanes and splitter vanes to re-unite in the test room. Secondary air for cooling the exhaust flow enters by way of two ducts, also equipped with splitter vanes, located below the primary intake and a third duct above the control and test room. These lead to the chamber housing the large diffuser into which the jet unit discharges. Some air is entrained in the diffuser and the remainder flows over the diffuser to join the gases escaping from a drum-type perforated spreader at the tail end. Then the diluted gases enter two cranked passages with vertical splitter vanes and are finally turned by horizontal cascade vanes to the vertical exhaust stack. Roofs and walls are lined with felted mineral wool secured by corrugated light-alloy sheeting.

Since the noise level from the efflux of jet engines often approaches 120 dB its analysis and reduction become of great importance. Investigations are currently being carried out in the industry with equipment made by the Danish firm, Bruel & Kjaer, whose products are handled in the U.K. by Rocke International Limited. The noise is picked up by a condenser microphone and fed to an audio-frequency spectrum



A. Bridge from control room.
B. Cold lock.
C. Pressure lock.
D. Thermal insulation.
E. Test chamber.
F. Observation windows.
G. Break joint.
H. Great door.
J. Sealing pad.
K. Great door carriage.
L. Reinforced concrete beam.
M. Concrete floor.
N. Foundation bed.
O. Insulating cellar.
P. Insulating support block.
Q. Fan motors.
R. Vacuum pipe.
S. Brick piers.
T. Dry air inlet pipe.
U. Cooling and turning vanes.
V. Air circulation duct.
W. Expansion bellows.
X. Air circulation fans.
Y. Chamber support.
Z. Access to windows.

Fig. 9 : 11—Vickers high-altitude test chamber.

recorder which not only analyses the frequency/dB relationship but also automatically records this information on a pre-printed chart. The frequency range covered is from 35 c/s to 18 kc/s and the microphone is capable of dealing with sound pressures up to 150 dB.

SIMULATED HIGH-ALTITUDE TESTING

The efficiency of gas turbines, and also the characteristics of metals and materials, being so markedly affected by the low temperatures experienced at high altitudes, it was to be expected that, quite early, the demand should arise for test facilities in which appropriate temperatures and pressures could be provided continuously. At the end of the War the only full-scale plant in existence was the B.M.W. 'Herbitus' chamber at Oberweisenfeld in Austria.

The layout of the piping associated with the plant is shown diagrammatically in Fig. 9 : 10 (top) and below is a chart indicating the corresponding pressure and temperature variations of the air as it passed through the plant. A three-stage centrifugal compressor delivered air at about 2·5 atm abs, and expansion down to desired altitude conditions took place in a single-stage axial turbine, the corresponding temperature drop being controlled by two large water and brine coolers forming part of the plant. Power recovered by the turbine was returned to the compressor so that the driving motor of the latter had only to make good the circuit losses.

The exhaust system comprised two four-stage centrifugal pumps which could be operated either in series or in parallel, depending on the altitude conditions required.

Illustrated in the centre of the diagram is the test chamber proper which consisted of a steel cylinder about 12 ft diameter and 30 ft long which could be evacuated to a static pressure equivalent to an altitude of 50,000 ft. Air could be supplied to the intakes at 550 m.p.h. and temperatures down to —65° C maintained. The rear end was detachable to enable the unit on test to be wheeled in or out and a special cylindrical nose-piece, which slid freely inside elaborate packing on the main intake nozzle, was provided. An air supply of 11 lb/sec under ground-level conditions kept the exterior of the unit cool and prevented overheating of the auxiliaries and electric cables. The recording room was entirely separate from the test house proper.

Such huge and elaborate plants cannot be built at short notice and early examples of both de Havilland and Rolls-Royce turbojets were flown out to the Herbitus chamber for altitude testing.

A somewhat more modest plant can provide high-altitude conditions for the static testing of components and materials. That built and operated by Vickers-Armstrongs at Weybridge is of the closed-circuit type with a working section 25 ft diameter and 50 ft long—large enough to accommodate the pressure cabin of a liner such as the Viscount. On each side of the working chamber is a pair of ducts, each 6 ft 9 in diameter, through which the air can be circulated at velocities up to about 70 m.p.h. by means of four fans each driven by a variable-speed 150 h.p. electric motor. One complete end of the chamber is retractable by means of a 40-ton hydraulic jack and can then be moved laterally to give open access for the loading or removal of equipment to be tested. At the other end are two air locks through which test personnel can enter. The structure is enclosed in a steel-framed building, the walls and roof of which are lined with a 12 in layer of insulating material.

For temperature control a refrigeration plant, comprising four 2-stage ammonia compressors each driven by a 150 h.p. electric motor, cools a supply of methanol which is circulated at a rate of 20,000 gal/hr through the 16 sets of copper turning vanes located in the return bends of the four air-circulation ducts. About 30 hours' operation is required to lower the air temperatures from + 15° C to —65° C. A 2-stage vacuum pump driven by a 140 h.p. motor lowers the air pressure to simulate, from + 15° C, an initial rate of climb of 1,500 ft/min to a ceiling of 80,000 ft. Pressure can be restored at rates up to the equivalent of a descent from 50,000 ft to sea level in 160 sec. An air-drying plant extracts moisture from all air admitted in order to prevent icing in the chamber at low temperatures.

The subject of icing in gas turbines, particularly those of pure jet type, is a matter of intensive research and some interesting information on the subject was given in a report issued by the National Research Council of Canada. Using a Junkers Jumo 004 axial-flow unit for test purposes, the icing was artificially induced by means of water-spray equipment. Most of the icing runs were made under simulated conditions at

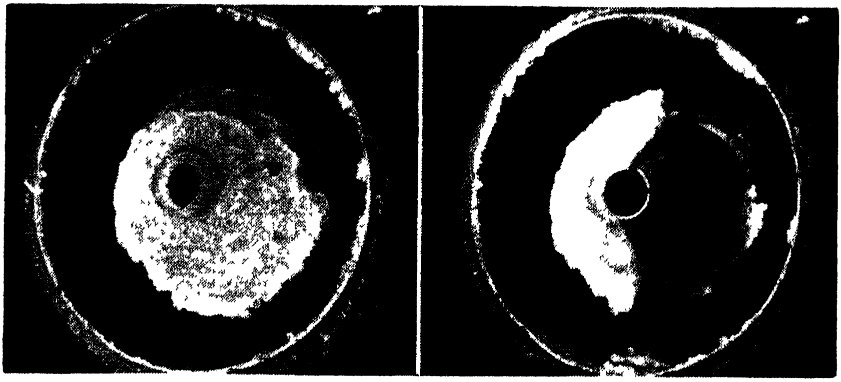


Fig. 9 : 12—Ice formation, intact and sectioned, on inner cone of Junkers Jumo intake duct after 22·5 min run. Air temperature — 2·8°C; water spray 21·9 lb; weight of ice cone 7·5 lb.

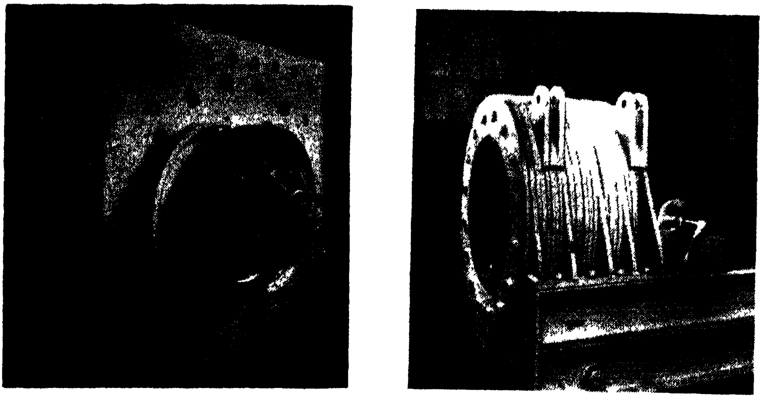


Fig. 9 : 13 Napier spinning rig with vacuum cover removed (left) and the retracted protection tunnel (right).

low ambient temperatures, the turbine speed being standardized at 7,600 r.p.m. As a result of these preliminary investigations, the general conclusions reported were that icing resulted in a drop in thrust and compressor performance, and an increase in exhaust gas temperature and fuel consumption.

SPINNING RIGS

Apart from the continuous research into the physical properties of materials there is need to study the behaviour of components when severely over-stressed. This applies with particular importance to rotating parts such as the discs and blades of compressors and turbines. For this purpose spinning rigs have been developed in which components can be rotated at speeds greater than those experienced in operation.

Non-destructive tests are made to determine effects of overspeeding and also, at strictly controlled speeds, for proof-testing such items as rotor discs before assembly in production units. At the Bristol plant data on the elastic deformation of a component under centrifugal stressing is obtained by an optical dilatometer. A beam of light is projected across the rim of a turbine disc, for example, and the shadow is magnified by an optical system and projected against a scale which can be read through a window in the cell.

For reasons of economy in running a cell is usually evacuated and thus the power required to drive the specimen has only to overcome mechanical friction. Since in the event of a fracture of the specimen fragments would be likely to be thrown off with the velocity of a projectile, substantial shielding by armour plate, torpedo netting, concrete or combinations of such materials is provided for the safety of operating personnel. In the case of research testing to destruction, the shielding is lined with heavy timber so that deformation of the detached fragments on impact, and consequent loss of evidential value, is obviated or at least minimized.

In the Napier rig for non-destructive testing the specimen is housed in a light alloy casing—relatively cheap to replace—spigot-mounted on a heavy armour-plate bulkhead and rotated by a 100 h.p. motor driving through a 20 : 1 increasing gear. Fixing bolts are proportioned to secure the casing rigidly during normal running but to shear in the event of a bearing seizure. The casing, slotted at its extreme diameter to allow the escape of any oil that may enter from the bearings, is completely enclosed in a domed vacuum cover clamped to the face of the bulkhead and making an air-tight joint with a rubber sealing ring.

Safety in operation is ensured by a protection tunnel built up of steel plate laminations, with intervening sheets of lead, drawn together by a series of 2 in diameter bolts. In the critical area the laminations are of armour plate but elsewhere they are of mild steel. The bore is lined with hard wood several inches thick. Mounted in a substantial carriage running on large-diameter flanged wheels on rails sunk in the concrete floor, the complete tunnel weighs more than 10 tons. It is moved up to position abutting the bulkhead, or retracted, by one man's effort through a simple train of high-reduction gearing.

FLIGHT TESTING

A most important part of the research, testing and development programme for an aircraft turbine power unit is conducted in the air under actual flying conditions. For this purpose the 'flying test bed' was developed from existing well-proved aircraft. Britain was fortunate in having available for this work in the early years the Avro Lancaster bomber and its variants the Lancastrian and Lincoln. These aircraft were particularly suitable for use as flying test beds for either turbojet or turboprop units, despite their modest altitude and air speed performance. Each had a fuselage sufficiently capacious to accommodate the necessary instrumentation panels, automatic photographic equipment, and a complement of test engineers.

With some of the smaller turboprops, such as the Rolls-Royce Dart and Armstrong Siddeley Mamba, the normal four R-R Merlin piston engines were retained, and the turbine unit to be tested was mounted in the nose. In the case of the larger gas turbines, however, the flight test practice was to install them in place of the Merlins in the outboard nacelles. The retention of the piston engines in the inboard plants gave an adequate safety margin during the exercise of deliberate tests of combustion system blow-out conditions. Representative British flying test bed installations were : Rolls-Royce Nene, Rolls-Royce Avon and de Havilland Ghost in the Lancastrian ; Armstrong Siddeley Mamba,

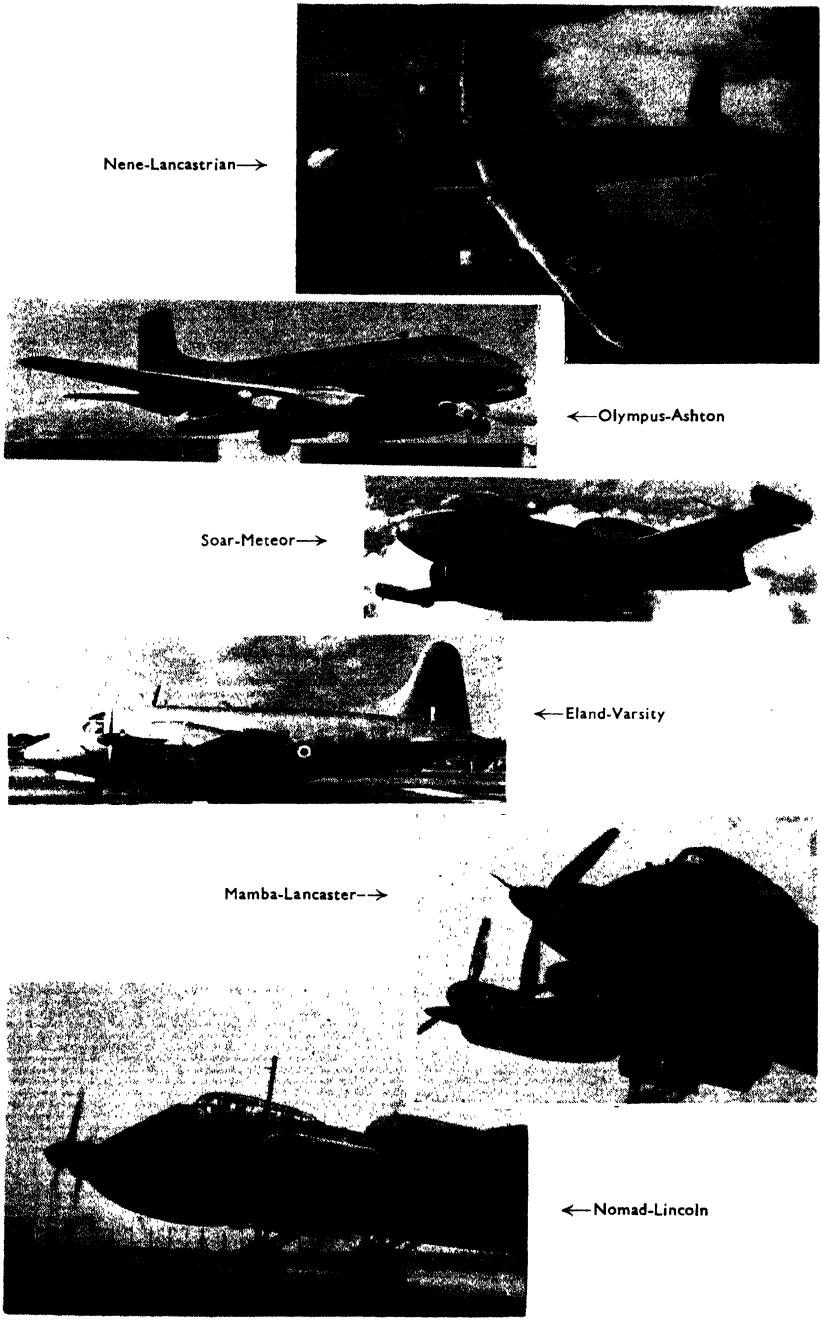


Fig. 9 : 14—Typical flying test beds.

Fig. 9 : 15—The Olympus-Canberra flying test bed took the world altitude record.

Armstrong Siddeley Python (see Fig. 4 : 15), and Rolls-Royce Dart in the Lancaster; Armstrong Siddeley Sapphire in the Hastings; and Bristol Theseus and Napier Nomad in the Lincoln.

Flight testing of new power units is also undertaken by substituting them for the standard units in production turbine aircraft, although these may not be able to offer all the facilities of a fully-equipped flying test bed or to accommodate a test engineer. Examples of this practice occur in the Rolls-Royce Trent-Meteor (see Fig. 4 : 7), Armstrong Siddeley Sapphire-Meteor, de Havilland Ghost-Vampire and Napier Eland-Varsity.

The war-time piston-engined bomber, however, is no longer adequate for testing purposes. Only a specifically designed turbine aircraft can utilize the power available and give the speed and altitude to test modern power units. The English Electric Canberra bomber is currently being widely used as a test vehicle. A typical example is the Bristol Olympus-Canberra which, it will be recalled, took the world altitude record in May 1953 at 63,668 ft.

For some investigations, it will be appreciated, even a flying test bed cannot provide room for all the specialized test equipment necessary for measuring and recording data during operation. However, actual measurements may be taken in the air and the information conveyed to a ground station for recording. The technique employed is an adaptation of that used for recording data transmitted from aircraft and guided missiles.

Rolls-Royce use this air-to-ground method to investigate the vibration of turbine blades. Strain gauges are secured to the turbine blades and the signals are taken to slip rings, arranged on the front end of the power unit, from which connections are made to the electronic transmission equipment. At the ground station, which is also connected by land lines to several static test beds, the signals are visually indicated by a trace on a cathode-ray oscillograph and recorded by a high-speed camera.

MAINTENANCE

A turbojet unit is much simpler than a piston engine by virtue of the smaller number of moving parts both in the main assembly and in the necessary auxiliary equipment. The absence of sliding parts such as pistons, piston rings and connecting-rod bearings ; highly stressed valves and their operating gear ; a supercharger and its gear drive ; the complete ignition apparatus ; and the airscrew and its reduction gear means the elimination of many items needing lubrication, attention and adjustment Lubrication of a jet unit is relatively simple, being confined to two or three main bearings and, of course, the auxiliary drives. Despite the much higher rotational speeds, the rotating assembly, due to careful design, high-quality materials, meticulous care in manufacture, assembly and balancing, and rigorous testing, is virtually trouble-free.

As a consequence of this simplification a considerable economy in maintenance is achieved as compared with that previously required for reciprocating engines This arises not only directly from the reduction of man-hours expended but indirectly from the smaller investment required for maintenance facilities and equipment, the fewer personnel necessary and, by the reduction of idle time, a quicker turn-round and higher aircraft utilization. The experience of the Royal Air Force in this respect is worthy of note. When in 1944 twin-jet fighters replaced single

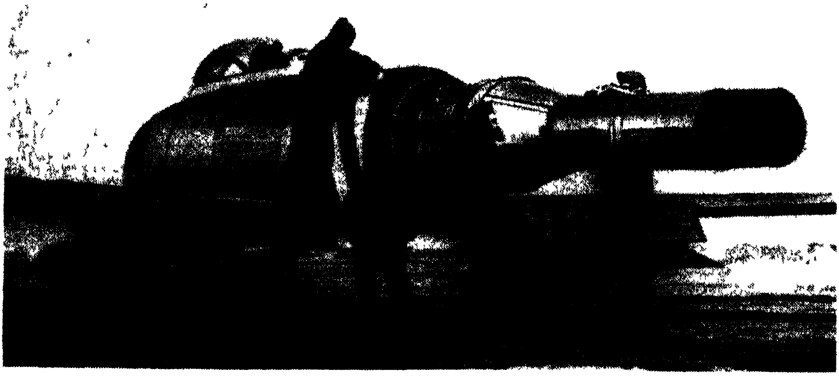


Fig. 9 : 16—Removal of the tail section of the F-86 Sabre exposes the cantilever-mounted jet unit for inspection, servicing or replacement.

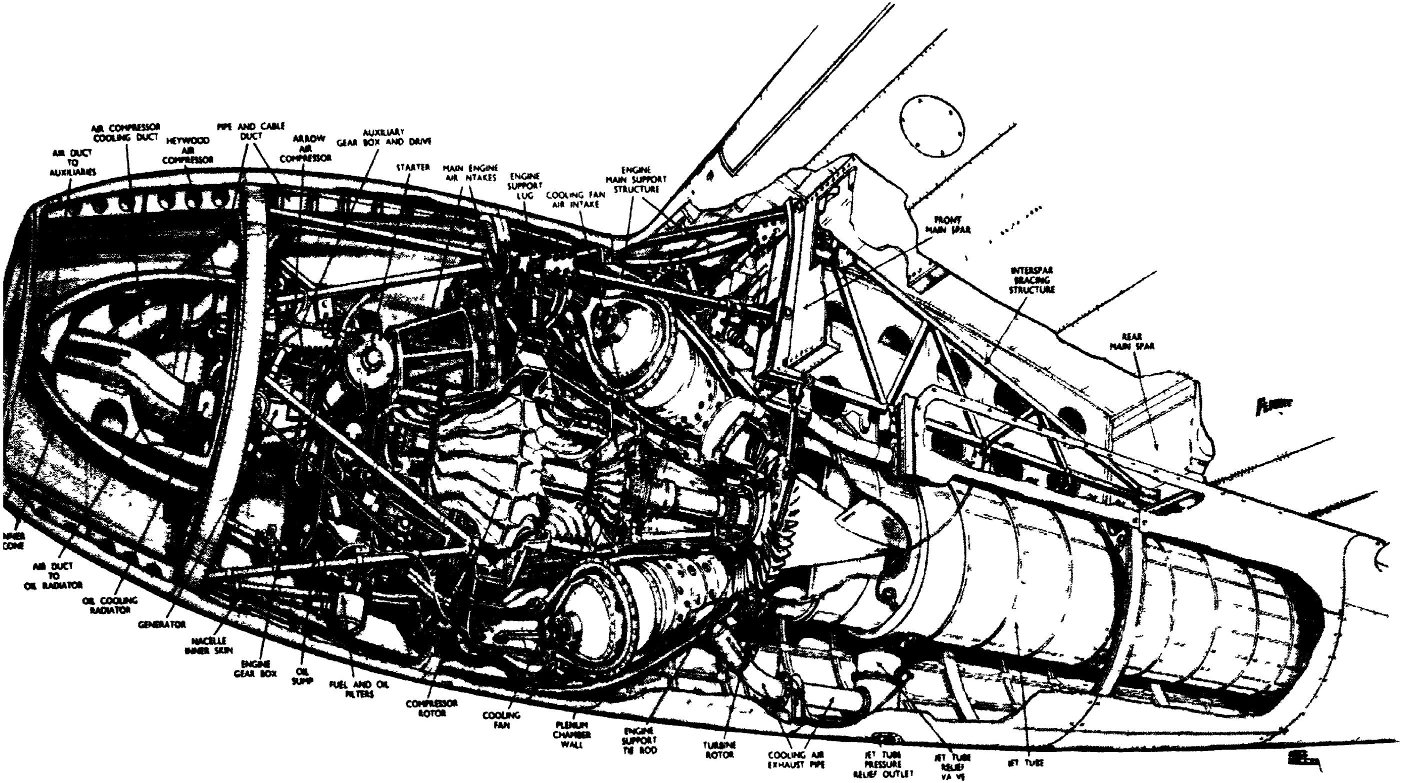


Fig. 9 : 17—Rolls-Royce Nene 2 installation in underslung wing nacelle on Lancastrian flying test bed.

piston-engined fighters it was found possible to reduce squadron maintenance personnel by 50 per cent.

The foregoing remarks also apply to the turboprop unit, but to a somewhat lower value due to the re-introduction of the airscrew and reduction gear. Scheduled regular maintenance is best compared on the basis of servicing man-hours per flying hour. A representative figure for a fully-developed piston engine rated for civil operation was 0·45. The experience of Rolls-Royce with the Derwent and de Havilland with the Goblin confirms that specific maintenance figures show a reduction of 50 per cent. With turboprop units the relative reduction, citing operation with the Armstrong Siddeley Mamba, Bristol Theseus and Rolls-Royce Dart, is about 33 per cent.

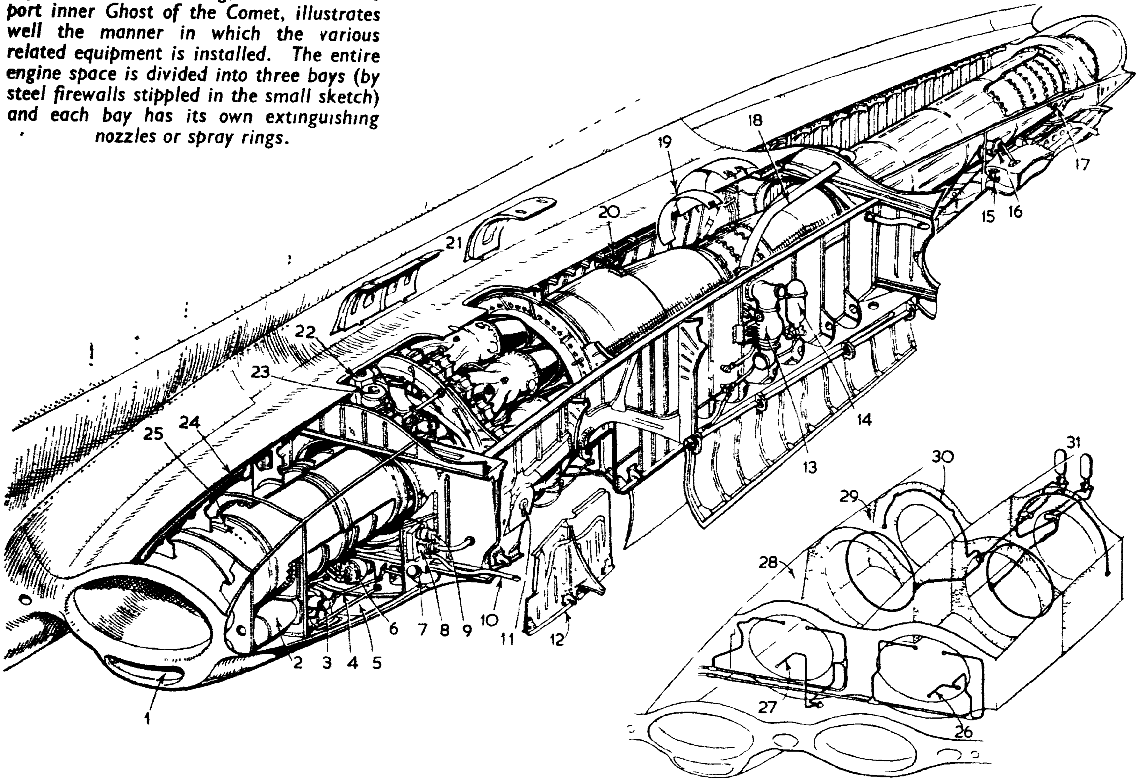
Turbine units are installed complete except for the tail pipe and, in the case of the turboprop, the airscrew. Invariably it is easier and quicker to change the entire unit than to attempt to deal with a major defect on an installed unit. Any attention to the rotating assembly would need a complete re-balancing which could not be effected whilst in the airframe.

Taking, for example, the Rolls-Royce Dart turboprop in the Vickers Viscount air liner, three fitters can install a unit in 40 min. The proving test, including start and runs at idle, cruise, climb, and maximum ratings and controls adjustment takes two men a period of 30 min. During the war period it was found possible for three fitters to install two Derwent turbojets in a Meteor and for the aircraft to be flying in 2 hr. A Ghost turbojet could be installed in the Comet in 30 min.

Scheduled maintenance follows a common pattern, of which that for the Rolls-Royce Dart in the Viscount is typical. Daily inspection and check occupies two men for an elapsed time of 20 min whilst a visual inspection between successive flights takes only 15 min by one man. After 50 hr running a more extensive check and inspection and cleaning of filters requires 1 hr 45 min work by two men. Every 250 hr a more elaborate inspection, including examination for evidence of flame tube distortion, is made by two men in 2 hr 15 min. At 500 hr the unit is opened up for internal inspection. This may provide the opportunity to renew the combustion chamber flame tubes, a straightforward operation that two men can complete in 3 hr 50 min. It could be made the occasion to remove the unit for overhaul, although this period is not firmly established and will certainly be extended.

Quite rightly, a conservative policy is adopted regarding period between overhauls. Special category tests on both turbojets and turboprops have shown that the 500 hr period can well be extended, particularly if a scheduled renewal of flame tubes is made. A de Havilland Goblin turbojet was run 1,000 hr on test cycles simulating operations on a fighter station with only 62 man-hr maintenance work, including two changes of flame tubes.

Fig. 9 : 18—This drawing, which shows the port inner Ghost of the Comet, illustrates well the manner in which the various related equipment is installed. The entire engine space is divided into three bays (by steel firewalls stippled in the small sketch) and each bay has its own extinguishing nozzles or spray rings.



1 Cabin-air heat-exchanger intake.
2 Cabin-air heat exchanger intake duct.
3 Hot air to heat exchanger.
4 Shutter control.
5 Detachable bottom panel.
6 Hydraulic pump.
7 Air supply duct.
8 Low-pressure fuel cock.
9 Fuel flowmeter transmitter
10 Fuel pipe to outboard engine.
11 Engine trunnion mounting location.
12 Access panel in engine rib.
13 De-icing air valves.
14 Methyl bromide extinguisher bottle.
15 Inner (split) flap centre hinge.
16 Flap-operating cable. wheel and linkage.
17 Tailpipe supporting roller.
18 De-icing air duct.
19 Detachable lagging.
20 Tailcone sling.
21 Access panels.
22 Alternator.
23 Starter.
24 De-icing duct.
25 Cooling air from alternators and rectifiers.
26 De-icing mixing chamber spray.
27 Wing equipment-bay spray.
28 Fire zone No. 1.
29 Fire zone No 2.
30 Engine spray rings.
31 Methyl bromide extinguisher bottles.

In 1953 the R.A.F. conducted a test of intensive flying with three English Electric Canberra bombers powered by Rolls-Royce Avon turbojets. Of the six power units, two were stripped for examination at 400 hr and one at 450 hr, and as their condition was completely satisfactory it was decided to continue the three remaining units to a total of 600 hr. Fifty flying days were taken by the first of these to complete 600 hr operation ; an average utilization of 12 hr per day although the peak utilization was 73 hr flying in a period of 84 hr. As the trial progressed the turn-round time was reduced from 1 hr 35 min to 36 min. At the end of the test the thrust output was within 3 per cent of the standard figure and the increase in specific fuel consumption was only 0·11 per cent. After examination the following parts were rejected as unserviceable : 4 compressor blades, 2 auxiliary drive roller bearings and seals, 2 flexible pipes, and 2 compressor shaft bolts. The flame tubes were only slightly affected and were accepted for a further service life of 400 hr.

A period of 600 hr running between overhauls represents some 300,000 miles flown in a jet aircraft and compares well with the 100,000 miles worked between overhauls by the rail locomotive or the commercial road vehicle.

As regards the actual work involved in an overhaul the turbine, again by virtue of its relative simplicity, shows to advantage. A civil-type piston engine of 1,500 h.p. requires about 500 man-hr for a complete overhaul, yet the figure for the Rolls-Royce Dart turboprop of 1,400 h.p. is only 350 man-hr. Comparison between a piston engine and a turbojet cannot be made directly but a relationship can be established for the purpose if it be remembered that at 375 m.p.h. 1 lb thrust equals 1 thrust horse-power (t.h.p.) and t.h.p. for the piston engine equals the b.h.p. times the airscrew efficiency. Established overhaul times for a 2,500 h.p. piston engine range from 750 to 850 man-hr. The Rolls-Royce Derwent, 3,500 lb thrust, needs 250 man-hr and the de Havilland Ghost, 5,000 lb thrust, only from 250 to 300 man-hr.

The quoted, and generally accepted, figures for the period between overhauls of piston engines are likely to be misleading. Due to their complexity and the multiplicity of parts it is not uncommon for individual units to be withdrawn prematurely. Thus, in a fleet of aircraft, it may be found that despite an official period of, say, 1,000 hr the average period is 850 to 875 hr. Given regular servicing, the reliability of the turbine unit should be maintained over its full rated period of operation and is expected to show a superiority to the piston engine in this respect.

CHAPTER 10

British and Canadian Gas Turbines

BACKED by unrivalled experience in the design and production of aircraft turbines, Great Britain stands pre-eminent in the quality and variety of her turbojets and turboprops. The current range presents very advanced ideas in design and manufacturing practice and some highly individual techniques, derived in some instances from the early researches of Whittle.

The individuality of British units, however, is not solely of a national character, but is expressed in the products of different firms. To some extent the peculiarities of British units can be traced to the urgency of military requirements during the war years and the necessity of avoiding speculative experimentation which might have interrupted the emergency developmental programme. It was this consideration that originally determined the use of centrifugal compressors and individual combustion chambers. While the theoretical superiority of the axial compressor or the annular chamber was not questioned, it was felt that lack of experience with such designs might have led to unwarranted delays in production. A brief account of early developments is given in the Appendix.

ARMSTRONG SIDDELEY

Python. Based on the background of experience with the experimental A.S.X. turbojet, the Python turboprop, of the direct-connected type having a 14-stage axial compressor, 11 combustion chambers and a two-stage turbine, was first run on test in March 1945. It earned the distinction of being the first turboprop to be ordered for production and served to power the Westland Wyvern naval strike fighter.

Air is taken in through an annulus located well back from the airscrew spinner and is divided between 11 throats in which the direction of flow is reversed. After passing forward through the compressor the air direction is again reversed in diffusing elbows before it reaches the combustion chambers arranged around the compressor casing. From the chambers the combustion gases flow through extension pipes, passing between the intake throats, to the turbine and thence by the tail pipe and a fixed area nozzle to atmosphere.

The main rotating drum of the compressor consists of two forged aluminium-alloy sections bolted together on the inside, the division being between the fifth and sixth stages. The outer casing carrying the stator blades is a light-alloy casting divided horizontally.

With a compression ratio of 5 : 1 the air mass flow at a maximum of 8,000 r.p.m. is 52·5 lb/sec. To facilitate starting by releasing a proportion of the air passing through the compressor, and also to assist acceleration after starting by preventing the stalling of the compressor blades, a blow-off valve is provided on the diffuser casing.

Drives to and from the compressor drum are by extension shafts of steel, bolted to the drum. This rotating assembly is carried in two angular-contact ball races at the front end and one ball and one roller bearing at the rear end. Both sets of bearings are cooled by air passing under the inner and over the outer races. The air for the turbine bearings is bled by external pipes from the fifth stage of the compressor, and that for the compressor bearings from holes in the compressor drum, at the seventh stage. The single turbine disc has a forked periphery to carry the two rows of fir-tree-rooted blades. Both front and rear faces of the disc are air-cooled.

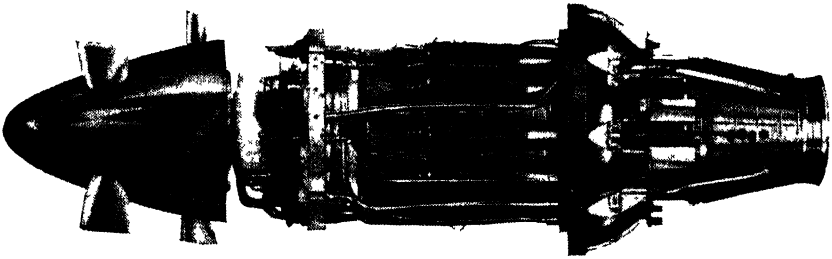


Fig. 10 : 1—Armstrong Siddeley Python turboprop, 4,110 e.h.p.

In addition to providing counter rotation, the airscrew gearing gives an overall reduction of 8·13 : 1. A propeller braking system is incorporated to stop rotation quickly when the fuel is cut off and also to prevent windmilling when the aircraft is parked. Take-off power at sea level at maximum r.p.m. is 3,670 s.h.p. plus 1,180 lb thrust giving a total of 4,110 e.h.p. At this régime the rate of fuel consumption is 415 gal/hr.

Mamba. Unlike the Python, the lower-powered Mamba turboprop has virtually a ‘straight-through’ air-flow system. Its diameter (29 in measured over the cowling) is remarkably small and the frontal area only about 30 per cent of that of an equivalent reciprocating engine. A 500-hr sealed test run of a standard Mamba was successfully completed in 1948.

The Mamba has a ten-stage axial-flow compressor which supplies six combustion chambers and a two-stage turbine. Air passes from an annular intake around the spinner into the compressor, the exit from which lines up with the combustion chamber manifold. Diffusing is started at the seventh compressor stage, and the airflow begins to be

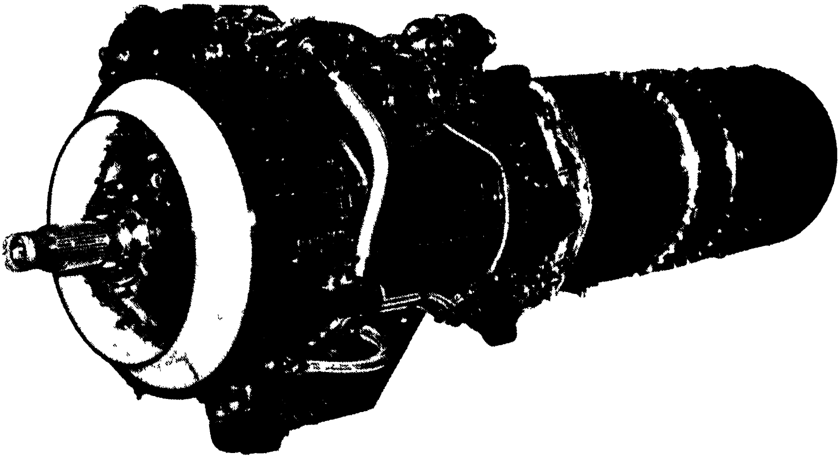


Fig. 10 : 2 Armstrong Siddeley late-series Mamba turboprop, 1,475 e.h.p.

slowed down from about 450 to 300 ft/sec. The air mass flow is 18·5 lb/sec.

In construction, the Mamba combined drum-and-disc-type compressor is unique, each stage of blading being riveted between the rims of a pair of steel discs shrunk on to the inner rotor drum (made of either steel or duralumin). The pairs of steel discs give a triangulated support to the blades, and all are dogged together to help carry the mechanical drive through the rotor assembly. Stationary shroud rings were originally carried by the stator blades, but this gave inefficient sealing, and there are now machined rings, spigot located, carried between the pairs of steel discs and lodging under the blade roots.

The reduction gearing is remarkably compact. A torsion shaft transmits power from the compressor front shaft to a helical sun gear. At its rear end, the torsion shaft carries six semi-circular splines—a design employed for ease of production ; at the front end, the sun gear is cone-mouthed and engages with the shaft (silver-plated, like the other fretting surfaces) through a number of small serrations. The sun gear drives three helical satellites and, attached one to each of them by means of screwed pegs, there are three spur planet gears. Completing the train is a large internally toothed annulus, stationary but free to float. The airscrew shaft and flange engage with the satellite carrier, the drive being transmitted through a Hirth coupling.

A sectional drawing of the Mamba detailing the constructional materials is given in Chapter 8. The standard unit, Mark 504, is type-tested and cleared for 500 hr between overhauls. At its s.l.s. maximum output of 1,475 e.h.p. the fuel consumption rate is 147 gal/hr.

Development of the Mamba is continuing, and has already resulted in a number of more powerful and efficient engines. Apart from the fact that some of these engines incorporate annular chambers no details are available of the improvements which they represent over earlier types. The principal rating* for the annular Mamba series is ASM.6, and engines of this type have been flown in the company's Dakota test-bed up to 30,000 ft. A late-model Mamba powers the production Short Seamew anti-submarine aircraft of the R.A.F. Coastal Command and Fleet Air Arm.

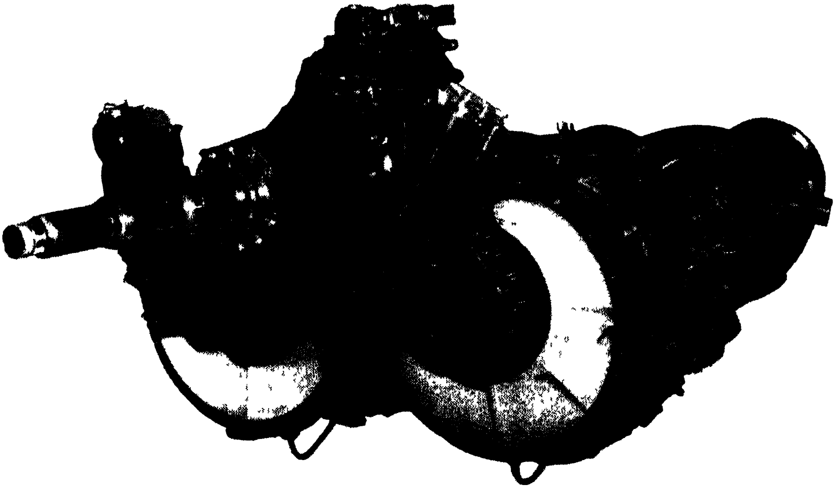


Fig. 10 : 3—Double Mamba for single- or twin-unit operation.

Double Mamba. The combination in a single power plant of two Mambas, each driving a co-axial, four-bladed airscrew in counter rotation, offers savings in weight and drag and provides 'twin-engined safety.' A common air intake and reduction gearing is used but each unit has independent fuel, lubrication and control systems. Both units are employed for take-off or emergency but cruising is done on one unit operating at optimum efficiency. As the majority of running is done on one unit, it becomes possible to extend both the period between overhauls and the total working life.

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* The term 'rating' is used in place of 'designation' because it is becoming a practice with the Ministry of Supply to indicate the acceptance of a new power rating by allocating a different number or letter to an engine which is in itself virtually identical with one or more of its predecessors.

The Double Mamba is in super-priority production for the Fairey Gannet for the Royal Navy. In this connection it is of interest to note it has been cleared for operation on Admiralty diesel oil instead of the usual kerosine.

The first production Double Mamba was the ASMD.1. Developing approximately 3,000 e.h.p., this engine utilizes power sections with separate combustion chambers housing a new type of flame-tube, easier to make than the original pattern and with a longer life. Each chamber houses four Armstrong Siddeley vaporizing burners, and very good results are now being obtained with all types of hydrocarbon fuel. Starting is effected by cartridges, and the slipstream from one airscrew is then used to windmill-start the other.

Considerable effort has been expended in developing the Double Mamba power control system in order to fit the engine for carrier work. In normal installations it is run at a constant speed of 15,000 r.p.m.; i.e. once this rotational speed has been reached in minimum airscrew pitch, extra power is added without increase in engine speed. Thus, very rapid build-up in power can be achieved when necessary. A Lockheed brake is fitted for bringing the engine to rest quickly after shut-down.

Later types of Double Mamba are under development, to give both greater power and a lower specific fuel consumption. It is expected that these engines will incorporate power sections with annular combustion chambers, as do the latest single Mambas. The ASMD.4 is already rated at an output of the order of 4,000 e.h.p.

Adder. A limited number of turbojet adaptations of the Mamba have been produced for specific applications for experimental and research duties. Thrust developed is 1,050 lb and it has been used to power the Australian Pika pilotless aircraft and also the Swedish SAAB 210 delta-wing research craft.

Viper. Several versions of this relatively simple, lightweight turbojet have been developed. It is designed for rapid production and in its 'short-life' form, ASV.3, makes the widest possible use of economical materials. Vipers were first used to power the Australian Jindivik radio-controlled target aircraft. Considering the maximum thrust is 1,640 lb s.l.s., the dry weight of 365 lb is exceptionally low and gives a weight : power ratio of 0·22 lb/lb thrust. The overall diameter is 23·25 in. Other

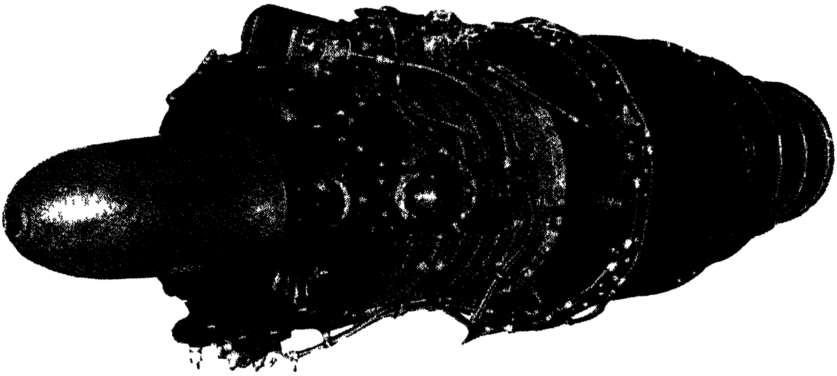


Fig. 10 : 5 Viper lightweight turbojet.

types in this category, ASV.4 and ASV.6, give thrusts of 1,750 lb and 1,900 lb respectively.

A 'long-life' version, ASV.5, intended for the Percival Jet Provost trainer is being developed for maximum reliability in service and a long working period between overhauls. It is conservatively rated at 1,640 lb thrust. A licence to manufacture the Viper in France has been taken up by the Marcel Dassault concern. In the Folland Midge, the Viper is the smallest engine ever to have exceeded the speed of sound.

Sapphire. In accordance with a Ministry of Supply decision on policy, the firm of Metropolitan-Vickers terminated their interest in aircraft power units in order to concentrate on gas turbines in other spheres. A large turbojet of Metrovick design, the Sapphire, intended for aircraft capable of transonic speeds, was transferred to Armstrong Siddeley for development. In November 1950 two units rated at 7,200 lb thrust were flight-

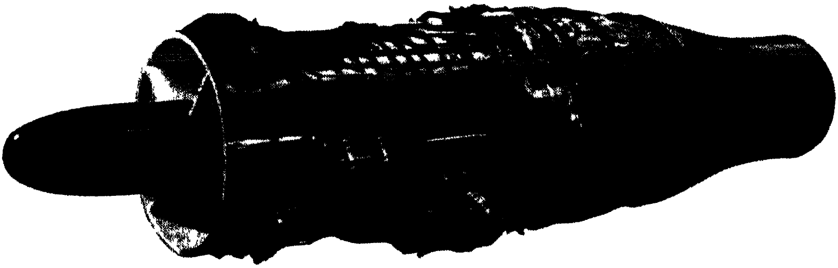


Fig. 10 : 6—Armstrong Siddeley Sapphire ASSa.6 rated at 8,300 lb thrust.

tested in a Hastings and early in the following year Wright Aeronautical Corp. acquired a licence to manufacture, together with other Armstrong Siddeley units, in the U.S.A. There the Sapphire has been developed as the Wright J65.

A Gloster Meteor powered by two Sapphires in August 1951 established four time-to-altitude records from a standing start. They were for 3,000, 6,000, 9,000 and 12,000 metres in 76, 110, 145 and 187 sec respectively. Since then the Sapphire has passed successfully a 150 hr type test at 8,300 lb thrust and has been ordered in quantity to power the Gloster Javelin all-weather, long-range, delta-winged fighter for the R.A.F. The specific fuel consumption is 0·9 lb/hr/lb at maximum power rating and 0 85 lb/hr/lb at cruising rating.

At the time of writing constructional details are withheld. It has been reported, however, that the compressor has 13 stages and is driven by a two-stage turbine. Externally it can be observed that the annular combustion chamber is equipped with 36 burners.

The two principal production types of Sapphire are those with M.o.S. ratings ASSa.6 and ASSa.7. The former engine completed a 150 hr type test at a thrust of 8,300 lb, and large numbers have now been delivered, both from the Armstrong Siddeley company and also from the Hawker Siddeley turbojet factory of Brockworth Engineering, Ltd. Sapphires are in R.A.F. service in such aircraft as Hunter 2s and 5s and in Javelin all-weather fighters. The ASSa.7 is a forward development with a more efficient compressor designed to handle a greater mass flow. Although no larger, and only slightly heavier, than earlier Sapphires, this engine has been type tested at 10,200 lb static thrust. Even more remarkable, this thrust has not involved any penalty in specific fuel consumption, for the figure recorded is below 0·9—a typical value being 0·89 lb/hr/lb.

Trials have been proceeding with after-burning Sapphires in a Canberra flying test bed. Such devices are in widespread use in the U.S.A., where after-burning American-built Sapphires power the F-104 and F11F fighters.

BLACKBURN-TURBOMÉCA

Blackburn and General Aircraft, Ltd., makers of the Cirrus range of piston engines for light aircraft, entered the gas turbine field during 1953 by acquiring the manufacturing and selling rights in the United Kingdom and the majority of the Commonwealth countries for the French-designed Turboméca series of units. The Blackburn-Turboméca Division at Brough, Yorkshire, is now established as a gas turbine development and manufacturing organization. While the British-built power units are still basically the same as their original French counterparts, externally and in respect of many details, they have been extensively re-designed.

The Palas turbojet of just under 400 lb thrust forms the basis of all the current units. It comprises a compact centrifugal compressor/annular combustion chamber/single turbine assembly, which may be regarded as the common gas producer of all the units.

Examples of Blackburn-Turboméca products in addition to the Palas are the Turmo, which has an output of 450 h p. at the shaft, which is at the rear end of this engine; another shaft turbine, the Artouste, of similar power, but with the drive-shaft in front, and the Palouste air compressor with an output of 2 725 lb/sec at 41 55 lb/in^2 at 15° C.

The interesting and unusual feature of these gas turbines is the interchangeability of their main assemblies. For example, the whole reduction gear at the rear end of the Turmo is identical with that used at the front of the Artouste The auxiliary drive-cum-air intake unit is applicable to Palas, Turmo and Palouste. Within the gas-producing section of this unit the compressors, turbines, combustion chambers and fuel systems are all interchangeable. The second stage of turbine employed in all but the Palas is also the same in each case, although it is used as a free power-turbine without mechanical connection with the gas producer section in the case of the Turmo.

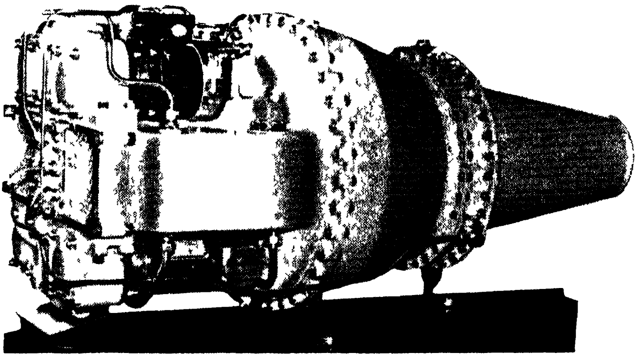


Fig. 10 · 7— The Blackburn-Turboméca version of the French-designed Palas turbojet, 390 lb thrust, weight 148 lb. The new intake and accessory arrangement is evident.

The four gas turbines named are each to be developed in two forms known as the 500 and 600 series. The difference is simply one of air mass-flow, which in the case of the 500 series is approximately two-thirds of that of the 600 series. This possibility originally arose out of the special requirements of the Palouste air compressor.

Blackburn also hold a licence to produce the Turboméca Marboré turbojet which may be loosely described as a scaled-up Palas. It produces 883 lb of thrust for take-off.

Turboméca units have a unique fuel system. A single pump and metering control feed the fuel to what is called a rotary atomizer mounted on the main shaft between compressor and turbine. The fuel flows via the shaft itself and is flung into the surrounding annular combustion chamber from perforations in the atomizer.

A number of primary power source applications for Turboméca units are planned in both fixed-wing machines and helicopters, and they are also to be employed as auxiliary power units for aircraft, naval and industrial purposes. In addition, the Turmo is to be developed as a twin or coupled unit with—in the first place—helicopter applications in view.

BRISTOL

Proteus 1. Before the design of the Theseus was completed (see Appendix) studies were commenced on a high-compression turboprop unit without heat exchanger. A ratio of 10 : 1 was to be obtained by using an axial compressor with blow-off valves, followed by two centrifugal stages. This conception was found difficult to realize within the dimensional restrictions of an aircraft unit and quite early in the development the second centrifugal stage was discarded. Furthermore, critical speed problems were encountered and rotational speed was curtailed. A compression ratio of only 5·6 : 1 was obtained and combustion chambers and turbine were redesigned to handle the increased volume flow. In this form it was intensively developed and culminated in the Proteus 2 units installed in the Bristol Britannia and the Saunders-Roe Princess.

Proteus 2. First run up on test in January 1947, the Proteus 2 is of the reverse-flow type with eight combustion chambers arranged around the centrally located compressor. Air enters by eight radial passages to the rear of the compressor and passes forward through the 12 axial stages and the final centrifugal stage to be reversed to the combustion chambers. As in the Theseus, the compressor is driven by a two-stage turbine and the airscrew by an independent single-stage turbine.

Maximum output, s.l.s., was 3,500 e.h.p. with a specific fuel consumption of 0·688 lb/e.h.p./hr. Cruising at 350 m.p.h. at 35,000 ft and assuming an airscrew efficiency of 80 per cent the consumption rate would be 0·510 lb/e.h.p./hr.

Coupled Proteus. On installations developed for wing mounting in the Saunders-Roe Princess flying boat, two Proteus units are arranged side by side, with their axes approximately 42 in apart, and are coupled by a double-helical gear train giving an initial reduction of 3·2 : 1. The

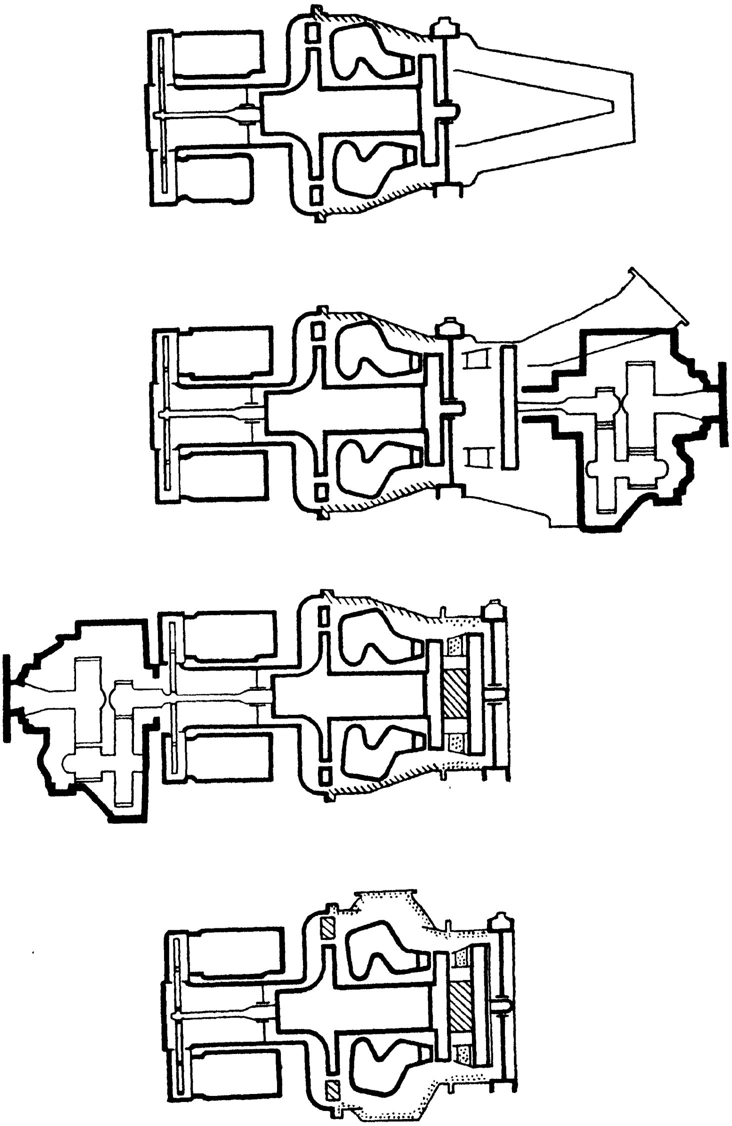


Fig. 10 : 8—Gas turbines in the Blackburn-Turboméca range are assembled from basic interchangeable units. From top to bottom, Palas, Turmo, Artouste, and Palouste (air compressor). The reduction gear used for the rear-drive Turmo is, for example, interchangeable with that for the front-shaft Artouste below it. The basic gas producer and accessory sections are common to all the engines.

counter-rotation gear for the eight-bladed 16 ft diameter airscrew has a ratio of 3·7 : 1 giving an overall reduction of 11·84 : 1 from turbine shafts to airscrews.

An important feature of the air intake system is the full use made of the airscrew slipstream ram. A pair of intake ducts is arranged (one duct each side of the main power cell) with their inlets in the leading edge, directly in line with the most effective part of the airscrew blades. These ducts extend to a point level with the compressor air inlets, where the flow is turned at right angles through the firewalls towards a plenum chamber completely encircling the compressor air inlets of both units.

Proteus 3. Of completely new design, with no part or dimension common to previous units, Proteus 3 derives directly from the experience gained in thousands of hours' running of Proteus 2. While retaining the basic reverse-flow layout, this more powerful unit, having a lower specific fuel consumption, is reduced in length and weight. The axial compressor has 12 stages, as before, but is of reduced diameter and now feeds directly into the single centrifugal stage instead of by way of a diffusing duct. Thus the length of the unit is materially reduced while the smaller diameter allows more space for the combustion system without adding significantly to the overall diameter. The turbine now has four stages, two for the compressor and two for the independent airscrew drive. To reduce leakage losses, tip shrouds have been provided on the blades of the first three stages. A 6 per cent gain in turbine efficiency was secured by this expedient, it is claimed.

Reference to the sectional drawing opposite reveals numerous interesting constructional features differing from earlier Proteus units. The steel ' corset ' which formerly extended from the mounting ring at the air intake to the diffuser casing and enshrouded the axial compressor has been discarded. With the reduced length and diameter, adequate rigidity is obtained by a double-walled compressor casing in RR 50 light alloy. The present mounting ring is located forward and supported by two coned rings of D.T.D. 184 bridging the diffuser casing.

As before, each of the three main component assemblies—turbine, compressor, and airscrew drive and reduction gear—is mounted for rotation as a separate unit with the necessary driving connections made by toothed couplings. Previously the compressor assembly was mounted on three anti-friction bearings, one at the rear and two forward, but the reduction in length has shortened the front overhang and made it possible to dispense with the front bearing. Since two bearings are much easier to mount in meticulously accurate alignment than three, a difficult production and assembly problem has been eased.

Another refinement is the provision of dowel bushes at the abutting faces of the light-alloy compressor rotor discs. Thus the eight through-bolts which draw up the 12 discs between the flanged forged-steel rotor ends are relieved of all shear stress. The rotor blades are forged in D.T.D. 282 stainless steel, which in practice has proved to be more consistently reliable than light alloy. Stator blades also are of forged stainless steel or alternatively precision cast in H.R. Crown Max. The compression ratio has been raised to 7 : 1.

Maximum output for take-off at sea level is 3,320 s.h.p. and 1,200 lb thrust, giving 3,780 e.h.p. Cruising at 300 knots (345 m.p.h.) at 35,000 ft, the specific fuel consumption is estimated at 0·495 lb/e h.p./hr.

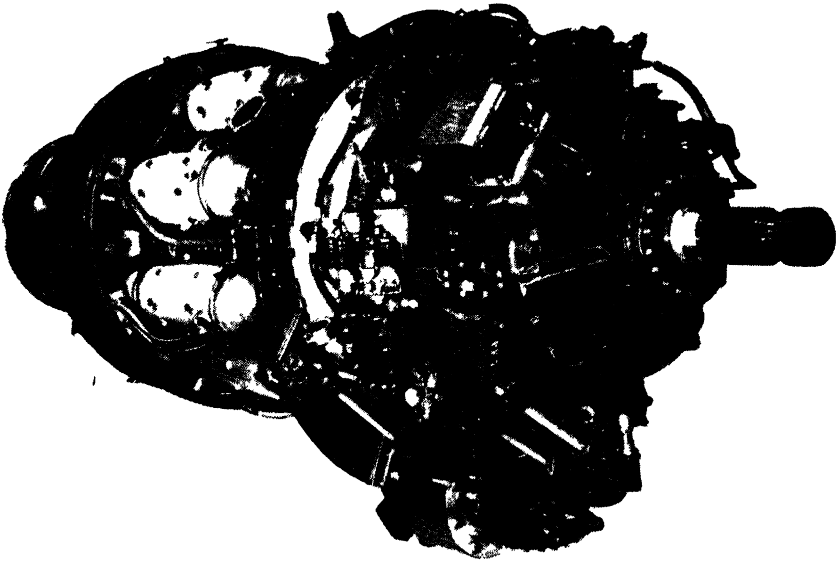


Fig. 10 : 10 Bristol Proteus 3 turboprop.

For the Series 300 Britannias, a more powerful Proteus has been developed, with the designation Proteus 750 *et seq.* Running at 12,000 compressor r.p.m., and with rather higher gas temperature, the 755 Proteus develops 3,650 s.h.p. plus 1,200 lb thrust, resulting in a total e.h.p. of 4,150. It incorporates the following features (most of which are standard on the earlier engine) : Ultra electronic control system which automatically selects the correct operating conditions and prevents excessive temperatures; double-helical reduction gear; full anti-icing; provision for bleeding off compressed air for cabin pressurization or accessory drives, and a highly sensitive overspeed control actuated by torquemeter pressure.

Olympus. Pursuing the policy of seeking efficiency by a high ratio of compression, Bristol has developed a turbojet unit which, even in its early versions, had a thrust of 9,750 lb and was the most powerful jet unit disclosed at that time. It is intended for large, long-range aircraft and accordingly is specially designed to secure a low rate of fuel consumption and a long working life. Overall dimensions have been kept to a minimum. Frontal area is approximately 8·7 ft^2, the overall diameter being 40 in.

Owing to the difficulty of obtaining flexibility in operation with a single axial compressor of wide ratio, resort is made to compounding by arranging low-pressure and high-pressure compressors in series. Each compressor is driven by an independent turbine through concentric shafts thus establishing the mechanical prerequisites that make possible the attainment of easy starting, efficient low-speed operation and rapid acceleration.

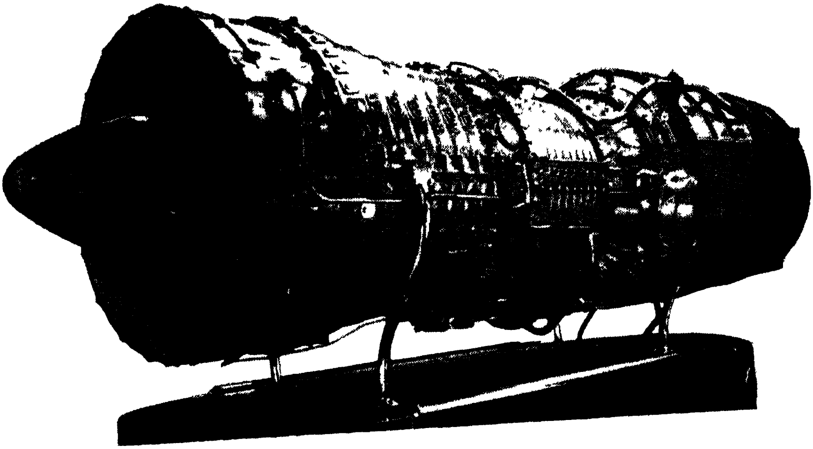


Fig. 10 : 11—Bristol Olympus two-spool turbojet.

At the time of writing details of construction were not available. From an external view, however, a six-stage l-p compressor, an eight-stage h-p compressor and an annular combustion chamber may be inferred. It will be noted that the guide vanes at the entry to the compressor are individually mounted. This enables their incidence to be adjusted on the development units but it is not expected that variable-incidence vanes will be a feature of the production version.

An early type gave a specific fuel consumption on test—presumably at maximum take-off power—of 0·766 lb/hr/lb. In October 1950, a few months after it was first demonstrated on the test bed, the Wright

Aeronautical Corp. acquired a licence to build the Olympus in the U.S.A. Two Olympus units powered the English Electric Canberra which on May 4, 1953, reached a height of 63,668 ft to establish a new world altitude record.

The first production Olympus was the series 101 which is produced for the Avro Vulcan with a type-tested thrust of no less than 11,000 lb. The Olympus 101 is a very fully equipped engine, yet it turns the scales at only 3,650 lb ; at its full thrust the specific fuel consumption is 0·79 lb/hr/lb, and this drops even lower—to 0·75—at 8,000 lb. The continuous cruising output is 9,600 lb, and the 101 can be accelerated from idling to full power in under 5 sec.

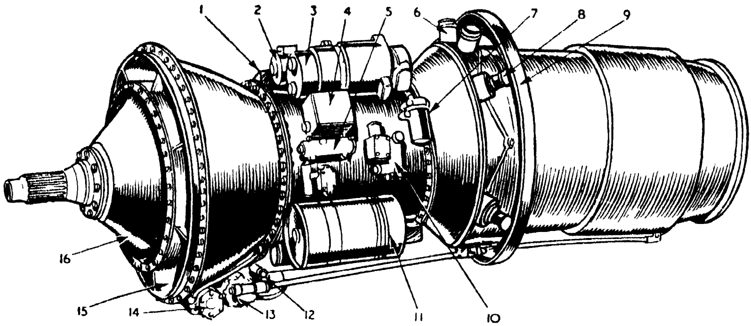
There are also more powerful types of Olympus under development to give powers suitable for the supersonic aircraft of the 1960 era. One of these is the BOl.6, which has already completed a considerable amount of bench running.

Orpheus. Specified for a very wide range of light attack aircraft, fighters, trainers and other types, the Orpheus turbojet is the only engine in its class in a comparable state of development. Termed a medium-thrust turbojet, it is a very neat single-shaft engine utilizing a compressor which is aerodynamically similar to the low-pressure compressor of the BE.25 (see below) but with aluminium, instead of steel, blades.

The thrust rating of the first types of Orpheus is 4,000 lb at 20,000 ft, and the weight is in the region of 850 lb. The engine may be equipped with a simplified type of re-heat designed by Bristol and weighing only 80 to 90 lb.

BE.25 All gas turbines so far developed have been ‘full-throttle’ engines, and they have accordingly lost power with increase in altitude. In the BE.25, the Bristol company have developed a turboprop in which the sea-level power is throttled to a maximum of about 4,000 s.h.p., or 5,750 e.h.p., by restricting the fuel flow. Not only does this provide all the power required for most of the immediate BE.25 applications, and without requiring the development of very large and critical gearboxes and airscrews—as needed on larger engines—but it also makes available full take-off power anywhere in the world, regardless of airfield height, humidity and temperature. Normally, the BE.25 can develop its full rated power up to about 20,000 ft, merely by opening the throttle.

The BE.25 is a two-spool, or split-compressor, engine with a high pressure ratio. One high-pressure turbine drives the high-pressure compressor and the three-stage low-pressure turbine drives the low-pressure compressor and airscrew. A typical calculated maximum-cruising rating at 350 kt at 36,000 ft is as follows : l-p compressor, 8,860 r.p.m.; h-p compressor, 8,700 r.p.m.; turbine entry temperature, 1,000° K ; mass



1. Saddle oil tank to starboard: 2. Electric starter: 3. 50 kVA alternator: 4. Ultra electronic control box: 5. Starting pump: 6. Main air bleeds: 7. Fuel filter: 8. Four dynamic suspension units: 9. Mounting ring: 10. Barometric pressure control, torque limiter and shut-off cock: 11. Oil cooler (port and starboard) fed by pipe from individual ram intake: 12. Oil-pressure transmitter: 13. Oil pump: 14. Scavenge filter: 15. Main air intake: 16. Airscrew reduction gear.

Fig. 10 : 12—Bristol BE.25 high-compression turboprop ; 4,000 s.h.p. from sea level to rated altitude of approximately 20,000 ft.

flow, 25·8 lb/sec; total power, 3,080 e.h.p.; fuel flow, 1,185 lb/hr; s.f.c., 0·385 lb/hr/e.h.p. The latter figure is considerably better than that possible with any piston engine in regular service with the exception of the Turbo-Compound which can match it in its best cruising condition. The BE.25 measures 41¾ in by 110 in and, although largely made of steel, is no heavier than the earlier-designed Proteus, at 3,058 lb.

DE HAVILLAND

Goblin. Designed to develop a thrust of 3,000 lb the Goblin 1 successfully passed its type test in January 1945 to gain the first type certificate issued for an aircraft propulsion turbine. It weighed 1,500 lb and the specific fuel consumption on test was 1·2 lb/hr/lb. Goblin 2, the early production version, developed 3,100 lb thrust at 10,200 r.p.m. and weighed 1,550 lb. It is officially approved for 600 hr operation between overhauls. Goblin 3, operating at the same gas temperature of 800° C at the turbine entry as in Goblin 1, had a maximum thrust of 3,350 lb at 10,750 r.p.m. and a fuel consumption rate reduced by 6 per cent. By 1952 Goblin 35 weighed 1,630 lb and was developing 3,500 lb thrust.

Apart from extensive contracts for the Royal Air Force, this remarkable power unit, installed in the Vampire fighter, has served to introduce and establish jet propulsion in more than a dozen countries.

During 1948 a Goblin was subjected to a test of unusual severity, constituting an equivalent of 462 operational flights by a fighter and entailing 500 hr running. Maintenance time was only 13·2 man-hr and 56 gal of oil were consumed. So encouraging were the results that the test was repeated, using the same Goblin after reconditioning at one-tenth of the cost of a new unit. On the second run 60 gal of oil were consumed as a slight leakage developed as the test progressed, and maintenance occupied 14·25 man-hr, plus 33·5 man-hr for a change of flame tubes. In normal service at that time flame tubes were replaced after 200 hr operation.

Basically, the Goblin comprises a single-sided centrifugal compressor which delivers air to 16 combustion chambers grouped symmetrically round the axis of the unit and leading to a single-stage turbine The compressor impeller and turbine rotor are connected by a tubular shaft to form a single rotating assembly mounted on only two bearings. It was fully realized that a double-sided impeller, as favoured by Whittle, appeared to make possible a unit of smaller overall diameter and that a large-diameter impeller, besides being more limited in rotational speed, would raise some problems in manufacture. Nevertheless, it was felt that there was a balance in favour of the single-sided design. The greatest possible use could be made of an uninterrupted airflow to the eye of the impeller through a large intake. Ram effect could be employed to the full, and eddy and friction losses were likely to be very small While a double-sided impeller made possible a compressor of small diameter for a given capacity, this advantage was to a great extent offset by the need to provide air passages around the outside of the compressor and between the combustion chambers for the supply of air to the rearward-

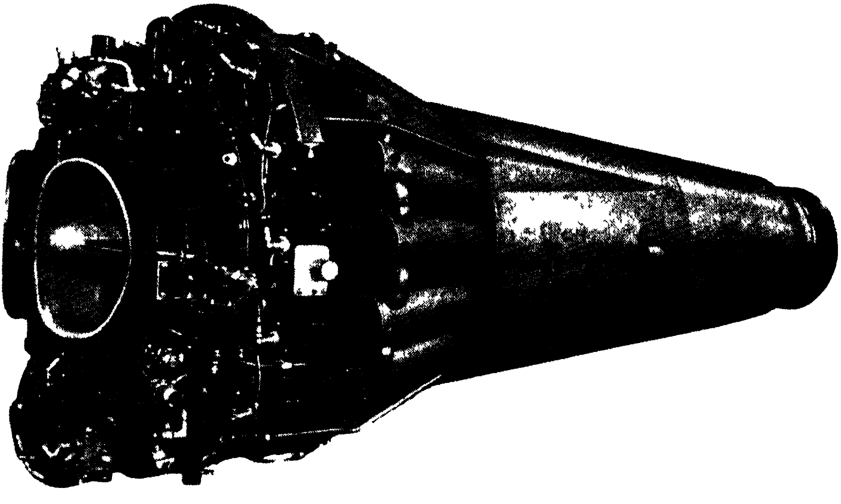


Fig. 10 : 13—De Havilland Goblin 35, 3,350 lb thrust.

facing intake, and by the less compact arrangement of the combustion chambers which would result. It was also argued that the rear intake, requiring a 180° turn in the airflow and likely to promote some degree of pre-heating, some turbulence and a lack of balance on the two sides of the compressor, would necessitate compromise in design tending to restrict breathing.

Two more advantages were foreseen for a unit with a single-sided compressor. First, the inherent simplicity and compactness would permit the use of only two bearings for the main rotating assembly, as compared with three required with a double-sided impeller. Secondly, the forward-acting thrust loading of the impeller would be almost sufficient to balance the axial load component on the turbine blades.

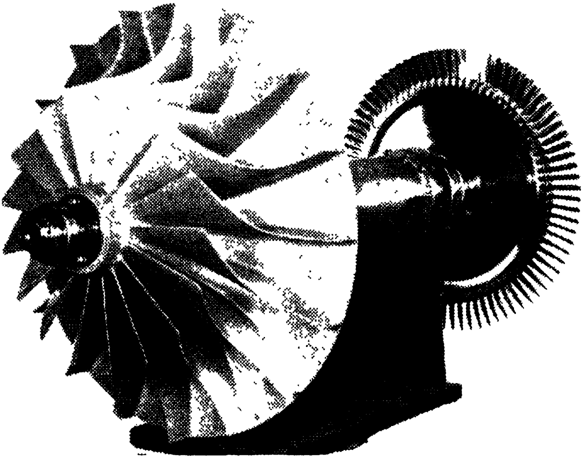


Fig. 10 : 14—The large single-sided impeller is a feature of the Goblin rotating assembly.

The Goblin impeller is a one-piece, heat-treated light-alloy forging which after being machined is anodized and polished. Furnished with 17 vanes, its diameter is 31 in and the peripheral velocity at 10,750 r.p.m. is 1,450 ft/sec. On the back face a series of concentric corrugations enter corresponding grooves in the rear casing plate to form a labyrinth-type seal. More than 6,000 h.p. is required to drive it at the maximum rated speed of 10,750 r.p.m. when the air through-put is 60 lb/sec. Pressure ratio is 3·67 : 1.

For the single-stage turbine the nozzle has 77 guide vanes and the rotor 83 blades, the numbers having no common denominator with a view to avoiding resonance. The fuel system is shown in Fig. 7 : 17.

Ghost. Following the successful development of the Goblin, the Ghost was originally conceived as a larger and more powerful unit of similar design for military applications. Two were built in 1945 and the first was run on test in September of that year. They had 14 instead of 16 combustion chambers and a maximum diameter of 57 in. The initial thrust of 4,500 lb was expected to be developed up to 5,000 lb.

The overall diameter was deemed to be too large and before the first units were completed the design was modified. The dimensional problem was solved by the introduction of cascade vanes to turn the air in the delivery ducts after leaving the diffuser. These enabled the diameter to be reduced to 53 in, which also led to a consequent reduction in weight. At the same time, in view of advances in the technique of combustion, a decision was taken to use ten combustion chambers of individually greater volume instead of 14. To avoid the use of splitter vanes in the diffuser, separate outlets were taken from each diffusing duct and two of these fed each combustion chamber. The redesigned unit was first run in October 1945 and developed its rated thrust of 5,000 lb early in 1946.

When the Comet airliner to cruise at high speed at 40,000 ft was projected, the Ghost was selected as the power unit. Thereafter, development was specially directed to questions of fuel consumption, reliability and safety in operation. A low specific consumption at high altitude calls for the highest ‘ installed efficiency ’ of the complete power plant and in this respect the Ghost, with its forward-facing intake, ducting air directly

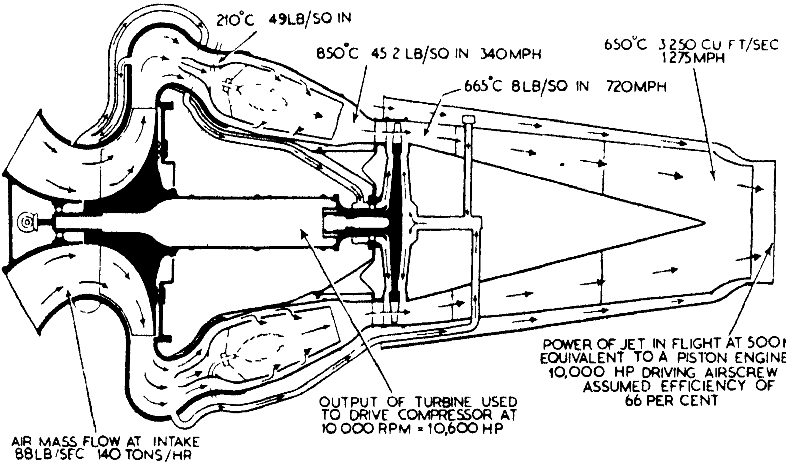

210°C 49LB/SQ IN
850°C 45 2 LB/SQ IN 340MPH
650°C 3250 CU FT/SEC 1275MPH
665°C 8LB/SQ IN 720MPH
POWER OF JET IN FLIGHT AT 500MPH EQUIVALENT TO A PISTON ENGINE OF 10,000 HP DRIVING AIRSCREW OF ASSUMED EFFICIENCY OF 66 PER CENT
OUTPUT OF TURBINE USED TO DRIVE COMPRESSOR AT 10 000 RPM = 10,600 HP
AIR MASS FLOW AT INTAKE 88LB/SEC 140 TONS/HR

Fig. 10 : 15—Diagram of Ghost gas-flow velocities, temperatures and pressures. Data refer to sea-level conditions.

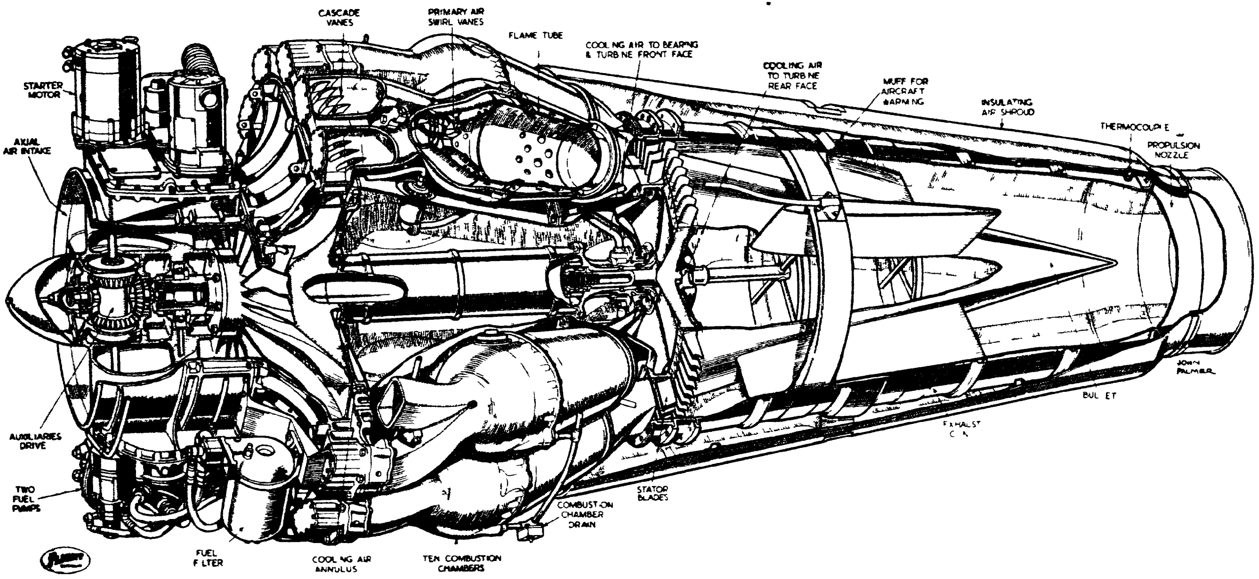


Fig. 10 : 16—A Ghost with axial intake, sectioned to reveal salient features Output is 5,000 lb thrust, s.l.s., at 10,000 r p m and weight 2,011 lb.

to the single-sided impeller, is well suited. A high duct efficiency raises the overall efficiency of the compression process and, in conjunction with favourable characteristics of the power unit itself, leads to high thrust and low consumption. In changing from military to civil versions of the unit many design modifications had to be made. The main compressors are tapped for cabin pressurization and also to provide air at 200° C which is piped to all aircraft surfaces requiring de-icing. Problems of temperature insulation and also of lowered power output arose from the buried installation in the wing. The inboard units required a tail cone, jet pipe and propulsion nozzle having an aggregate length of 19 ft. As a consequence the maximum thrust had to be raised to over 5,200 lb so that 5,000 lb thrust was available when installed. Water-methanol injection and afterburning equipment is available.

All the later types of Ghost (developed since 1951) are bifurcated-intake engines for Venom fighters. The Ghost 104 and 105 are the most important engines, and each has been delivered in large numbers. Strenuous efforts have been made to reduce weight and increase power, and considerable attention has also been paid to such problems as re-lighting at high altitude and providing for accessories—such as high-capacity alternators for radar. A good deal of experimental work has been carried out with afterburning, but no Ghost so equipped has entered service, apart from one type built by S.F.A. (Sweden).

Gyron. On July 30, 1953, the existence of a new turbojet, the H.4 Gyron, was officially admitted. It is stated to be of higher power output than any other unit in the world and that it will make possible the attainment of supersonic speeds in level flight by military aircraft. No details have been disclosed but speculative references have appeared in the American press. For what they are worth, these suggest a seven-stage axial compressor, the use of titanium alloy materials, the attainment of 16,000 lb thrust and the probable development to produce 20,000 lb thrust.

It is of interest to recall the 'not unduly optimistic' views expressed by the late Major F. B. Halford, formerly chairman and technical director of the de Havilland Engine Co. and the designer of the Goblin and Ghost, in a paper he read before the Royal Society of Arts on August 16, 1946. For a liner of 190,000 lb total weight flying non-stop from London to New York with 15,000 lb payload at 615 m.p.h. at 40,000 ft altitude he proposed four turbojets. These would be nominally rated at 12,000 lb thrust at 6,200 r.p.m. with 15,000 lb thrust for take-off. A 15-stage axial compressor would be driven by a three-stage turbine that at maximum speed s.l.s. would need to develop 28,500 h.p. Overall dimensions would be diameter 5 ft and length 17 ft and the weight 6,000 lb.

NAPIER

Eland. Additional to the Nomad compounded unit, described in Chapter 13, the Napier Co. is actively developing the Eland, a turboprop of the direct-connected type for use in conjunction with a single-rotation, constant-speed airscrew. In general layout it comprises a ten-stage axial compressor feeding six combustion chambers and driven by a three-stage turbine. The tubular braced structure extending from the mounting ring at the end of the compressor to the turbine casing, characteristic of the earlier Naiad unit, is retained but the subsidiary structure from the mounting ring to the reduction gear casing has been eliminated.

Throughout the design every effort has been made to keep down dimensions and weight. As a result the unit can be installed in a nacelle of the minimum dimensions and drag that can be accommodated behind a 3,000 h.p. airscrew and spinner. Overall dimensions are diameter 36·1 in and length 104 5 in and the dry weight is 1,575 lb.

Maximum output s.l.s. is 2,690 s.h.p. and 825 lb thrust or 3,000 e.h.p , giving a specific weight of 0·525 lb/e.h.p. Specific fuel consumption is 0 624 lb/e.h.p./hr at take-off power and 0 49 lb/e.h.p./hr when cruising at 350 m.p.h. at 30,000 ft.

The Eland is scheduled to power the Fairey Rotodyne, see Chapter 15. For this aircraft a special version of the engine has been developed in which an auxiliary compressor has been added behind the basic power section. During take-off of the Rotodyne all the power from the Eland is transmitted, via a hydraulic coupling, to this auxiliary axial compressor, which pumps atmospheric air up to propulsive jets at the tips of the hollow blades of the rotor. During transition to forward flight oil is drained from the transmission coupling until, eventually, all the power is being taken by the normal reduction gearbox and airscrew at the front. The auxiliary compressor then windmills under the influence of the air induced centrifugally through the rotor blades.

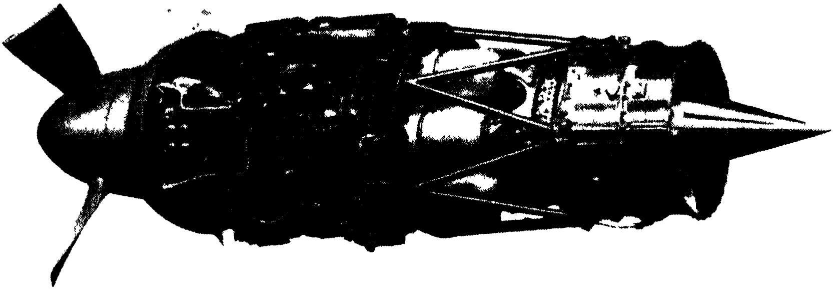


Fig. 10 : 17 – Napier Eland turboprop, 3,000 e.h.p.

Various types of Eland are under development and construction. Basic fixed-wing Elands have been installed in Varsity, Convair 340 and Ambassador test-beds, the last two aircraft also being equipped as furnished demonstration aircraft. Development is also rapidly proceeding upon a rather more advanced Eland capable of eventual rating at over 4,000 e.h.p.

Oryx. One of the smallest axial gas turbine engines ever built, the Napier Oryx exists in several forms for both fixed-wing aircraft and helicopters

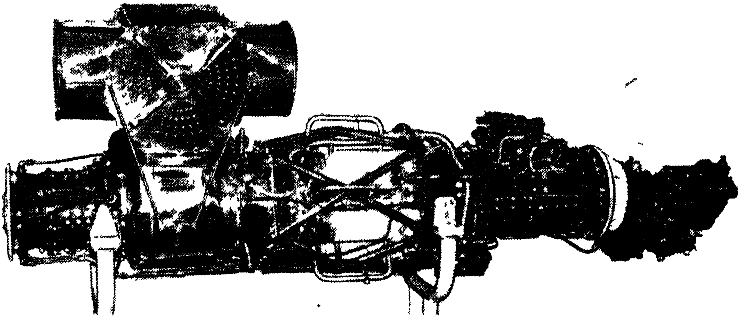


Fig. 10 . 18 Napier Oryx to produce 750 gas h.p.

Most Oryxes are arranged to deliver their shaft power not directly through a gearbox but to an auxiliary compressor. From this compressor air (or air and gas) is ducted to a remote turbine driving a power take-off shaft. Alternatively, as in the Percival P.84 helicopter, the gas delivery is ducted up to propulsive jets at the tips of the hollow rotor blades.

The Oryx is unconventional in appearance yet logical in layout. The overall dimensions are : length, 76 in and maximum diameter, 19¼ in ; the dry weight is 495 lb and the maximum gas horse-power 750, with a specific fuel consumption of 0·68 lb/hr/g.h.p.

ROLLS-ROYCE

Derwent. The first Derwent ran on the test bed in 1943 and in November that year passed a 100 hr type test at 2,000 lb thrust. Early the following year the Series 1 unit completed flight tests in the Gloster Meteor at the service rating of 1,800 lb thrust and weight of 920 lb. Intensive development followed, culminating in a successful 500 hr test under type-test conditions. Later types developed higher outputs, the Series 4 units being rated at 2,400 lb thrust.

The fruits of experience with the Derwent were embodied in the

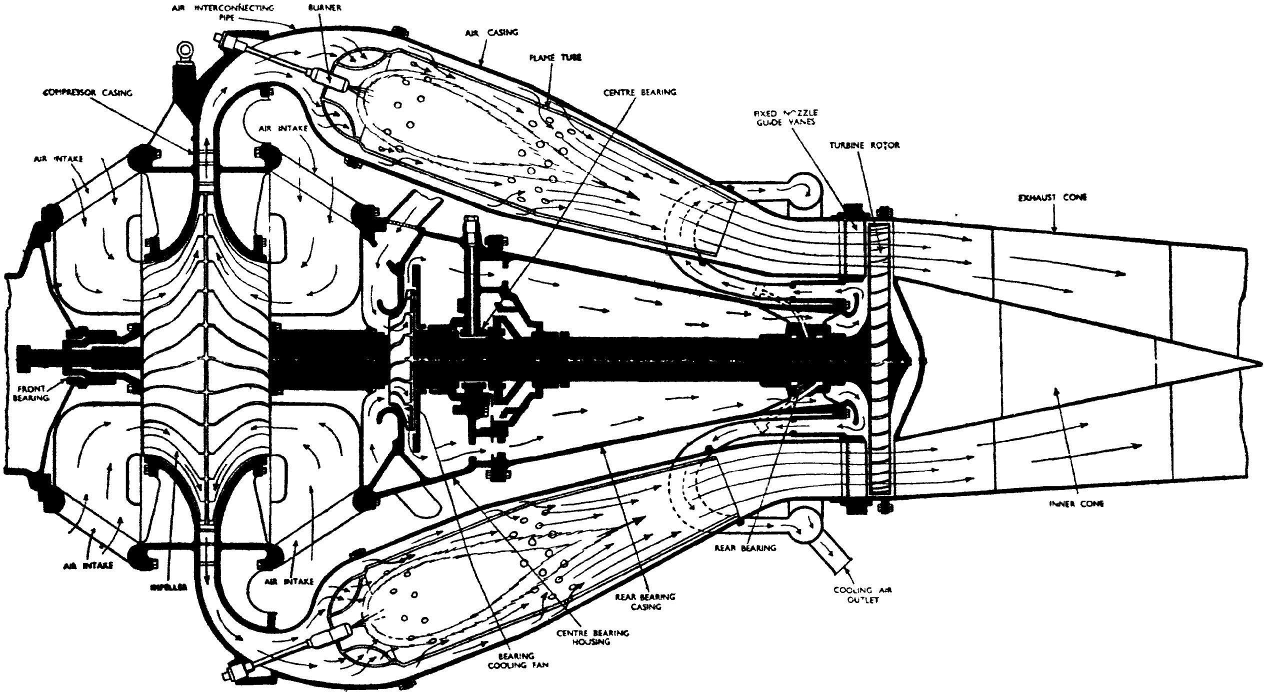


Fig. 10 : 19—Flow diagram of Rolls-Royce Derwent turbojet.

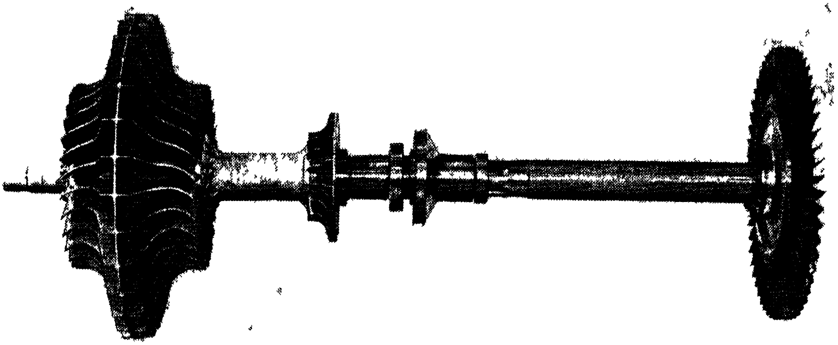


Fig. 10 : 20—Derwent rotating assembly. Left to right is double-sided impeller, cooling air blower, toothed shaft coupling and turbine rotor.

design of the larger and more powerful Nene turbojet built in 1944. Not unnaturally, the success of this new engine suggested a re-design of the Derwent and the Series 5 resulted. The first unit passed its 100 hr type test at an initial rating of 3,000 lb thrust This version early distinguished itself by powering the Gloster Meteor that in November 1945 established the world record speed of 606 m.p h. and in September of the following year raised the record to 616 m p h.

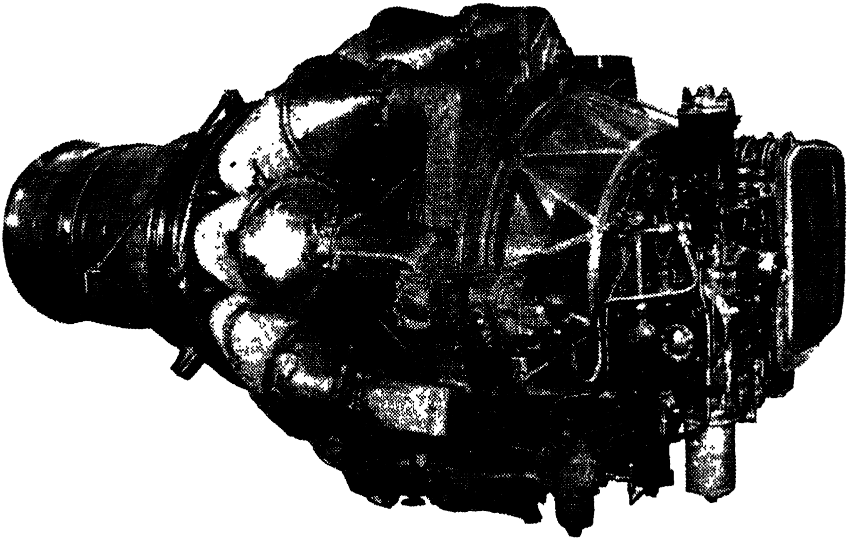


Fig. 10 : 21—Production version of Rolls-Royce Derwent 5, 3,500 lb thrust.

In effect a scaled-down version of the Nene, the Derwent 5 embodies the characteristic Rolls-Royce double-entry centrifugal compressor, with an impeller 20 68 in diameter having 29 radial blades on each side, and a single-stage turbine. The rotating assembly is mounted in two roller bearings and a central ball-thrust bearing Air is induced on both sides of the impeller, that for the rear inlet being fed past the necks leading from the diffuser to the nine individual combustion chambers. The earlier series had ten chambers

To cool the centre and rear bearings and the front face of the turbine disc a small centrifugal fan mounted in front of the centre bearing induces atmospheric air through short stub pipes and delivers it to the central casing. After passing up the face of the disc the cooling air is collected in a ring manifold and discharged to atmosphere. By employing this inexpensive blower to deliver relatively cool air at low pressure instead of tapping the main compressor, a saving of approximately 30 h.p. is effected.

The compressor impeller and turbine rotor, each with its shaft, are statically and dynamically balanced both individually and collectively as a single assembly The shafts are connected by a quickly-detachable toothed coupling, with a ball centre on the turbine shaft seating in the end of the impeller shaft. Should it be desired to remove the turbine rotor for inspection it is necessary merely to insert a hand through an access door in the casing, slide back a lock ring located by spring detents, withdraw axially the outer coupling element, turn the shaft about 30° to break the interlock and the rotor can then be withdrawn to the rear. Fifty-four blades with fir-tree roots are fitted to the periphery of the rotor, the tip diameter being 17 38 in.

Nene. The Nene was designed to meet a 1944 requirement for a turbojet to develop 4,000 lb static thrust while not exceeding a diameter of 55 in or a weight of 2,200 lb. In a remarkably short period—less than six months—the design was completed, all drawings prepared, the first unit built and a proving run of 1 hr successfully accomplished in October 1944. With a maximum diameter of 49·5 in and a dry weight of 1,600 lb, the designed performance was well exceeded and the Nene was soon cleared for flight at a maximum thrust of 5,000 lb at 12,400 r.p.m.

The double-sided, centrifugal compressor delivers air at four times atmospheric pressure to nine straight-flow combustion chambers. Measuring 28·5 in diameter, the impeller is machined with 29 radial vanes on each side from a single light-alloy forging. Curved entry vanes, approximately 17·75 in diameter, for each side are separate components, machined all over. At a 5,000 lb thrust rating the air mass flow is 89 lb/sec or 143 ton/hr.

A compressor having a double-sided impeller was chosen for the Nene

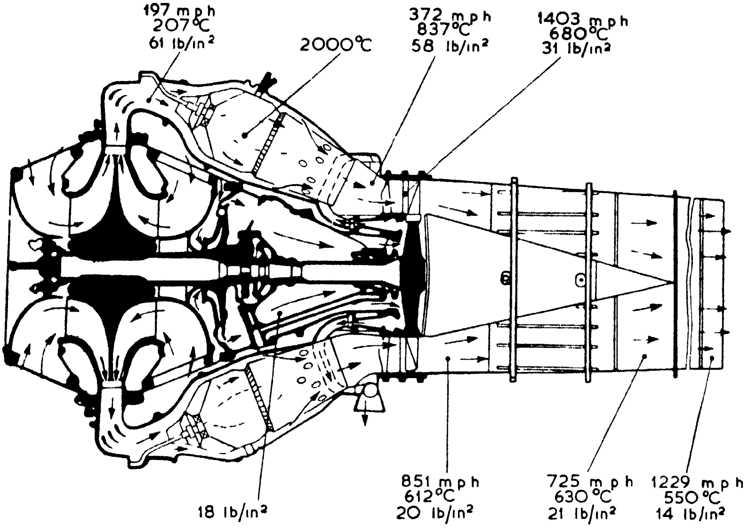


Fig. 10 : 22 – Velocities, temperatures and pressures (abs) in Rolls-Royce Nene.

because output of a turbojet is largely determined by the amount of air consumed, and this is conditioned by diameter of the compressor entry. Obviously two intakes can admit more air than one. Conversely, for a given quantity of air, the overall diameter of the impeller and consequently the complete unit can be relatively smaller than in a single-sided design. The wing nacelle or fuselage compartment forms a plenum chamber from which air is drawn to the compressor intakes. Velocity is lowered and any object sucked in may well fall to the bottom of the enclosure instead of passing into the compressor.

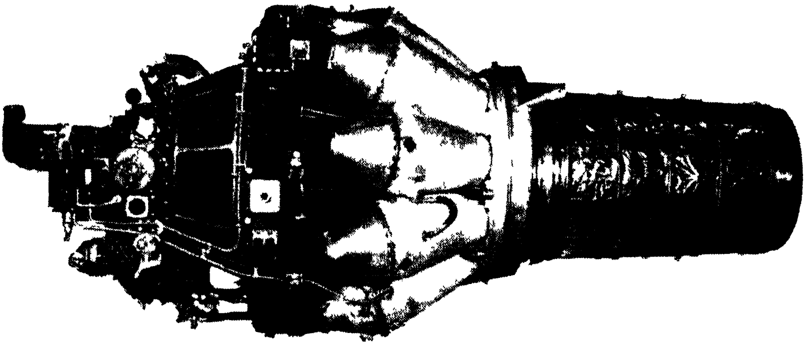


Fig. 10 : 23—Rolls-Royce Nene RN.3, 5,100 lb thrust.

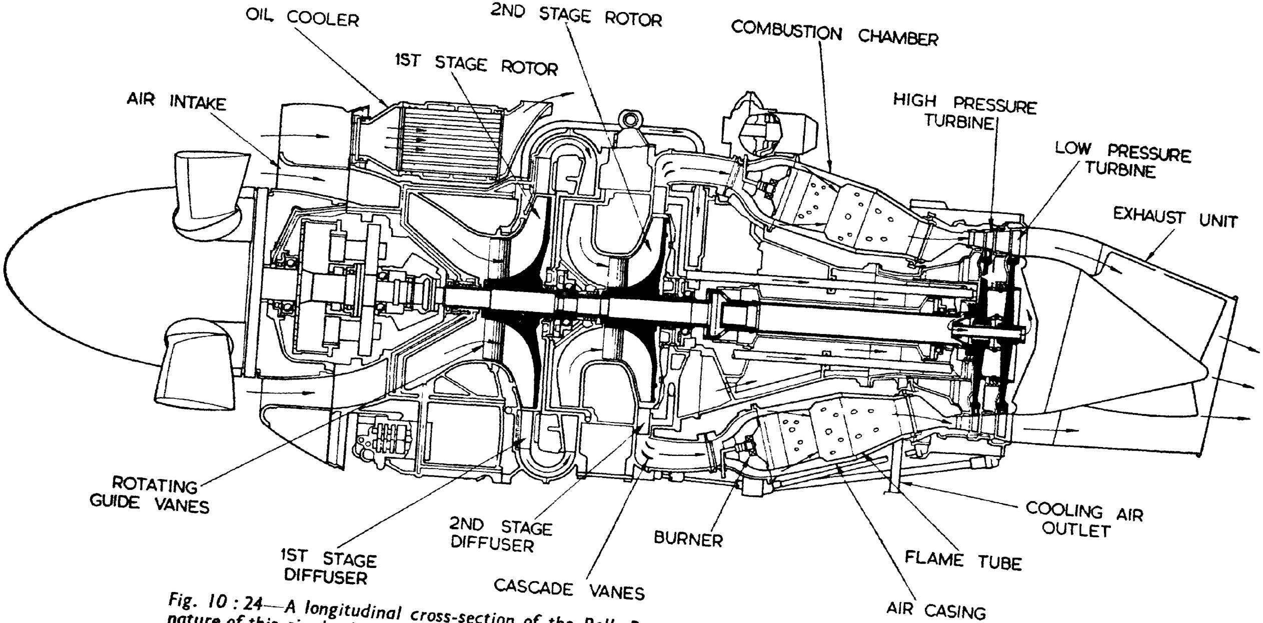


Fig. 10 : 24—A longitudinal cross-section of the Rolls-Royce Dart 505 turboprop. The essentially simple nature of this single-shaft engine is apparent ; the discs of the compressor and turbine stages are shown in solid black.

On the Nene the compressor casing is built up of front and rear members attached to a central diffuser ring by bolts passing through the diffuser vanes and the intermediate splitter vanes. To facings at the nine outlets from the diffuser ring are bolted the cast elbows conducting the air to the combustion chambers. In the bend of each elbow are three cascade vanes formed of lengths cut from an extruded section and cast in position. A pair of trunnions providing the main supports for the complete unit are also mounted on the diffuser ring.

The basic production version, Nene 2, is rated at 5,000 lb thrust at 12,300 r.p.m. and weighs 1,550 lb. Type RN.3, which has delivered 5,500 lb thrust on the test bed, is conservatively rated at 5,100 lb at 12,500 r.p.m. and has a weight of 1,620 lb. Nenes are in production for the Hawker Sea Hawk and Vickers-Supermarine Attacker and are also built under licence overseas, notably by Pratt & Whitney in the U.S.A., and Hispano in France. The Nene has also been built in Canada, and a number of developments of it have been manufactured on a huge scale in the Soviet Union.

Dart. Intended as a 1,000 s.h.p. unit for either civil or military use, the Dart turboprop was first flight tested in a Lancaster in October 1947. At the beginning of 1949 it was type-tested and given a rating of 1,250 s.h.p. plus 300 lb thrust and a few months later it was prepared for a further test for a rating of 1,400 s.h.p. plus 295 lb thrust.

In 1951 it successfully completed a 500 hr test under conditions simulating those likely to be experienced by Viscount liners in service on European routes. Only routine maintenance, amounting to 9·2 man-hr per 100 hr running time, was required. This production version had a maximum output of 1,400 s.h.p. and 365 lb thrust giving 1,540 e.h.p for take-off.

Of the ‘ direct-connected ’ type, a two-stage turbine drives a two-stage centrifugal compressor and the airscrew. The rotating assembly follows Rolls-Royce practice as turbine wheels and compressor impellers are mounted on separate shafts connected by a toothed coupling. Each single-sided impeller has 19 radial vanes and the curved entry vanes are of steel. The compression ratio is 5·5 : 1 and at 14,500 r.p.m. maximum speed the mass air flow is 18 lb/sec. From the compressor the air is distributed through elbows to the seven combustion chambers which are mounted helically. This arrangement avoids the use of right-angled bends in the air flow path and also slightly reduces overall length.

The turbine rotor is an assembly of two wheels on a common shaft. First-stage wheel has 123 blades on a disc of Jessops 18B steel and second-stage has 103 blades on a S.62 stainless steel disc. All rotor blades are of Nimonic 80 alloy and attached by fir-tree roots.

A double-reduction gear having a helical high-speed train and a final spur gear drive to the airscrew shaft has a ratio of 10 99 : 1. The annular air intake duct enveloping the reduction gear is in turn enshrouded by the lubricating oil tank.

Dart flying time in commercial service far exceeded 250,000 hr in the early part of 1955, and new Darts are going into service all over the world (in Viscounts) at a current rate of between 16 and 24 per month Very considerable detail modification has taken place to improve overhaul life (by early 1955 this was up to 1,000 hr) and both power and efficiency. The original production engine, the Dart 505, has now been succeeded by the Dart 506 and 510, both of which are listed in the table on p. 239.

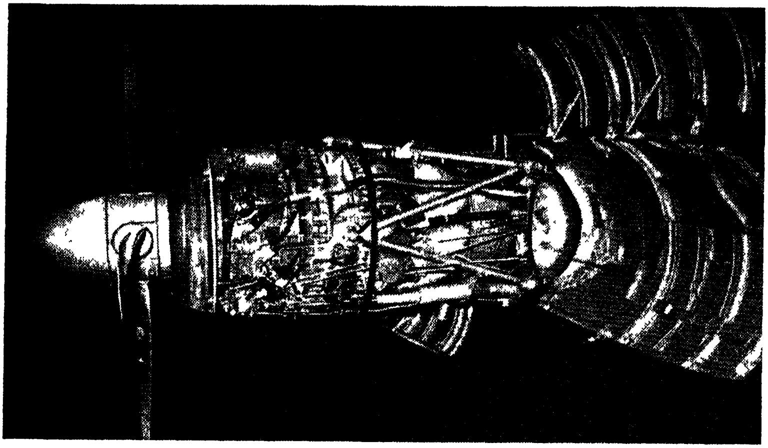
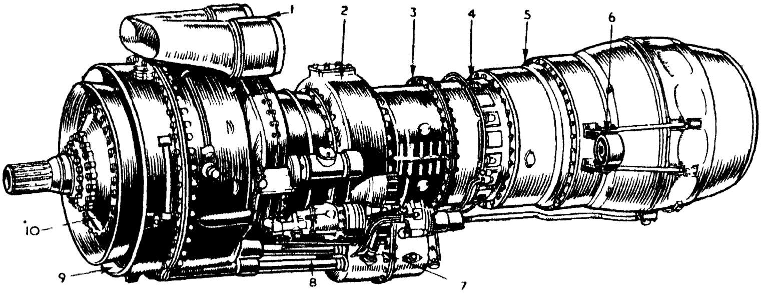


Fig. 10 : 25—Rolls-Royce Dart installed in a Vickers Viscount airliner. With the special cowlings folded back, the engine is unusually accessible.

R.B. 109. Benefiting from their extensive turboprop experience Rolls-Royce have designed a completely new turboprop with this designation. From front to rear, the R.B.109 has : a direct annular intake surrounding the reduction gear, a circumferential oil tank with twin oil coolers on top, a tapering low-pressure axial compressor, a circumferential duct for bleeding off air, a small-diameter high-pressure compressor, a very compact combustion system (clearly of the annular or cannular type) and a multi-stage turbine. Accessories are mounted around the lower flanks of the compressor casing.

As at present conceived, the engine is a full-throttle unit. Diameter and length are, respectively, about 32 in and 115 in, and the dry weight is around 2,000 lb. With a maximum sea-level mass flow of 40 lb/sec



1 Twin oil-coolers mounted above oil tank 2 Compressor bleed face fed from annular duct 3 End of rear compressor and beginning of steel section 4 Duplex fuel gallery and 10 burners 5 Beginning of turbine section 6 Shock-absorbing mounting 7 Engine accessories 8 Accessory-drive shaft 9. Main air intake 10. Reduction gear and torquemeter housing.

Fig. 10 : 26 —R.B.109 ; 4,000 s.h.p. for take-off.

(twice that of the Dart) the R.B.109 delivers 4,020 s.h.p. plus 1,000 lb thrust, resulting in a total e.h p. of 4,350.

Avon. An indication of the importance attached officially to the Avon turbojet is that seven years after it was first exhibited at the Farnborough Show in 1948 details of design and construction were still restricted. In this period it received intensive development, achieved some remarkable performances in the latest aircraft and was placed in production on the widest scale. Apart from the parent company plants, Bristol, Napier and the Standard Motor Co. are also building Avons. Overseas licences to produce in Australia, France and Sweden have been arranged and others are likely to be granted.

When first announced in 1948 the thrust rating of the version designated RA.2 was 6,000 lb and the weight 2,240 lb. Progressively since then the rating has been raised and, significantly, the weight : power ratio (lb/lb thrust) has been lowered. As the unit is intended for installa-

Fig. 10 : 27—Rolls-Royce Avon RA.7, 7,500 lb thrust.

tion in fighters, bombers, naval and also civil aircraft, a number of specific types have been developed in various power ratings.

By 1951 the RA.3 weighing 2,440 lb was type-tested at a rating of 6,500 lb thrust. Early in 1952 the RA.7 unit with a rating of 7,500 lb thrust was type-tested. This weighed 2,460 lb, giving a specific weight of 0 33 lb/lb thrust. The addition of afterburning equipment increased the thrust by 27 per cent from 7,500 lb to 9,500 lb and the RA.7R completed a 150 hr test at this rating in January 1953.

Although internal construction has not yet been disclosed it has been reported—but not confirmed—that the compressor has steel blades throughout 12 stages and that the turbine is a two-stage unit. Externally four blow-off valves, to ease starting and to relieve the compressor at high altitudes, are to be seen. An unusual feature is the provision of variable-incidence entry vanes to the compressor. These, 36 in number, are operated by a pneumatic actuator. A cartridge-type starter is housed in the central nose fairing at the air intake.

The first basic family of Avons to go into production started life at 6,500 lb rating, and many hundreds of engines at this rating power Canberra B.2s. These engines have given excellent service, and have stripped so well after periods of arduous service exceeding 600 hr that the take-off and maximum-power rating has been increased to over 7,000 lb. From the basic RA.7 has sprung another wide range of engines, those for the Mks 2 and 4 Hunters being rated at upwards of 7,250 lb and the civil Series 502 or 503 engine giving 7,150 lb for an overhaul life which should rapidly be established at 1,000 hr or more.

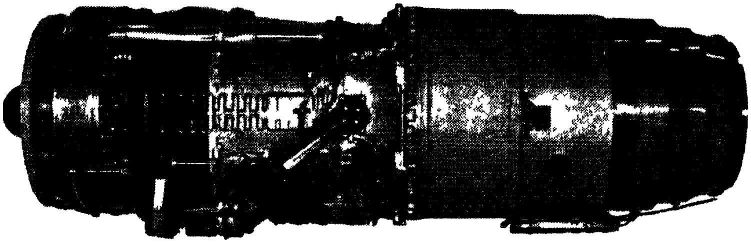


Fig. 10 : 28—Rolls-Royce Avon RA.28 of over 10,000 lb thrust.

In 1951-2 a considerably redesigned family of Avons was prepared, with a more efficient compressor of greater pressure ratio, an annular combustion system capable of accepting a greater mass flow although having a lower overall diameter, and widespread detail engineering changes. The first of this new range of engines was the 9,500 lb thrust RA.14. First production batches of these later Avons are for the Valiant medium bomber, in which they are believed to have a five-figure thrust rating. A commercial engine, with the designation Avon 521, has been developed for such aircraft as the Comet 3 and Caravelle, and this is at present rated at 10,000 lb thrust with the prospect of 11,000 lb available if required.

Soar. The R.B.82 Soar is, in relation to its size and weight, by far the most powerful gas turbine in the world. An essentially simple single-shaft turbojet, the Soar has a small axial compressor, a high-intensity annular combustion system and a turbine which, for starting purposes, can be spun by an external compressed-air jet. The whole engine is contained within a ' monocoque ' shell which requires no outer fairing. Very advanced techniques have made possible the achievement of a type-tested thrust of 1,810 lb (1,860 lb has been recorded on early Soars) from a unit with a diameter of 15¾ in, a length of 62¾ in and a dry weight of 267 lb. On this basis the thrust/weight ratio is almost 7 lb per lb. At full power, the specific consumption is 1·26 lb/hr/lb. The Soar is in production and is to be used in a number of applications, of both long- and short-life nature, here and in the U.S.A.

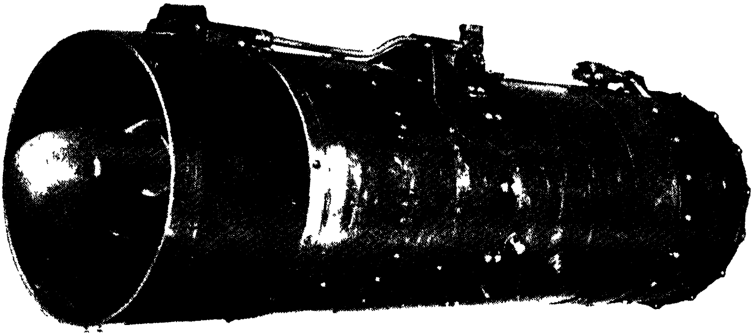


Fig. 10 : 29 —Rolls-Royce Soar RSr.2 turbojet ; 1,810 lb thrust. The diameter is 15¾ in and the weight 267 lb.

Conway. A unit of the by-pass type, the Conway has long been undergoing development and the existence of the RCo.1, RCo.2, RCo.3 and RCo.5 has been disclosed. The operating characteristics of a by-pass unit—relatively low fuel consumption and improved low-speed performance as compared with a pure jet—make it particularly suitable as a power unit for jet transports. The Conway is specified, it has been announced, for the Vickers 1000 transport which, as regards dimensions, is 40 per cent. larger than the Valiant bomber.

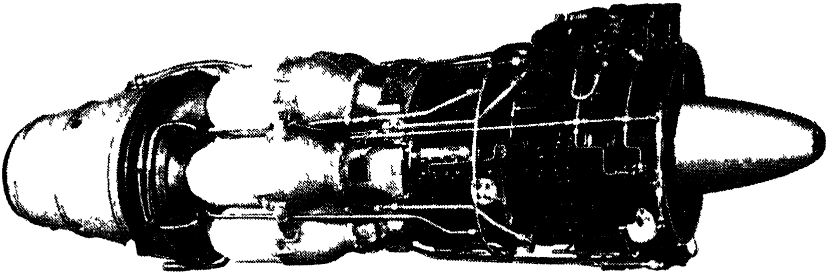


Fig 10 . 30 Avro Canada Chinook turbojet.

ORENDA ENGINES (FORMERLY AVRO CANADA)

Chinook. Canada's interest in aircraft turbines dates from the success of the early Whittle units during the war. The National Research Council sent a commission to England to study the new development and in 1944 promoted its own organization—Turbo Research Ltd.—in Canada to further studies and train personnel. In 1946 A. V. Roe (Canada) Ltd., a member of the Hawker Siddeley group, acquired the facilities of Turbo Research Ltd and set up a fully equipped research, test and development establishment at its plant at Malton, near Toronto.

Work was commenced on the design of an experimental pilot unit to develop a thrust of 2,600 lb as a prelude to a larger unit to be later placed in production. Named the Chinook, this unit was first run on test in March 1948. It has a nine-stage axial compressor of relatively advanced design giving a compression ratio of 4 5 : 1. Built up of light-alloy discs for the first eight stages and a stainless-steel disc for the ninth, the rotor is supported on a self-aligning roller bearing at the front and a double-row ball bearing in a self-aligning mounting at the rear. Compressor rotor blades are of stainless steel for the first and second stages with aluminium-alloy blades for the remainder.

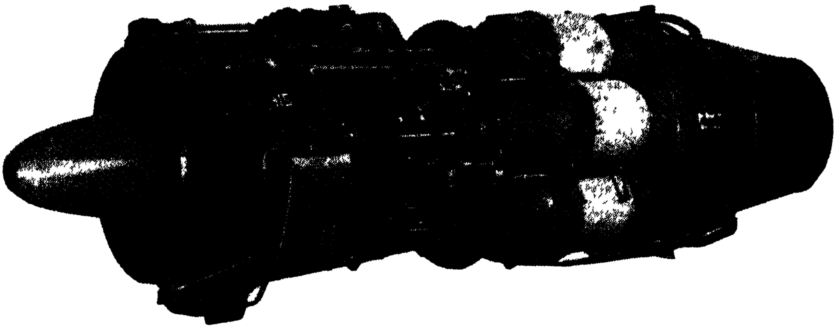


Fig. 10 : 31—Avro Canada Orenda, 7,000 lb thrust.

British and Canadian Aircraft Turbines

Manufacturer and Type	*Layout (see footnote)*	*Diameter (in)*	*Length (in)*	*Dry weight (lb)*	*Max. Power Rating (sea-level static)*				*Cruising s.f.c. (at altitude)*
					Thrust (lb)	*s.h.p.*	*r.p.m.*	*s.f.c.*	
Armstrong Siddeley									
Python	14, 11, 2	54·0	123·2	3,450	1,180	3,670	8,000	0·81	
Mamba 101	10, 6, 2	29·0	87·3	780	405	1,320	15,000	0·8	
Double M.1	2 (10, 6, 2)	44·0	101·8	2,100	810	2,640	15,000	0·8	
Double M.4	2 (—, Ann, —)	—	—	—	—	3,000	—	—	—
Sapphire 6	13, Ann, 2	35·0	129·0	2,550	8,300	—	8,600	0·89	—
Sapphire 7	—, Ann, —	37·4	127·6	2,972	10,200	—	—	0·885	—
Viper 101	7, Ann, 1	28·0	66·0	465	1,640	—	13,400	1·05	—
Viper 10	7, Ann, 1	—	—	—	2,300	—	—	—	—
Blackburn									
Palas 600	C, Ann, 1	17·1	25·32	148	390	—	34,000	1·2	1·14
Twin Turmo	2 (C, Ann, 1)	37·0	46·2	580	—	900	34,000	1·04	1·07
Turmo 600	C, Ann, 1	17·1	46·12	281	—	450	34,000	1·04	1·07
Bristol									
Proteus 705	12+C, 8, 2+2	39·5	113·0	2,850	1,200	3,320	11,700	0·64	0·495
Proteus 755	12+C, 8, 2+2	40·1	108·7	3,025	1,320	3,650	12,000	0·62	0·464
B.E. 25	—+—, Cann, 1+3	41·75	110·3	3,058	—	4,000	10,500	0·55	0·385
Olympus 101	—+—, Cann, —+—	40·0	120·0	3,650	11,000	—	8,500	0·79	0·75
Orpheus	7, Cann, 1	32·0	97·0	750	4,000	—	10,500	1·1	—
de Havilland									
Goblin 35	C, 16, 1	49·8	100·5	1,629	3,500	—	10,750	1·14	
Ghost 103	C, 10, 1	53·0	130·5	2,175	4,850	—	10,250	1·09	
Gyron	—, —, —	—	—	—	15,000+	—	—	—	—
Napier									
Eland	10, 6, 3	36·0	104·2	1,661	825	2,690	12,500	0·624	0·48
Eland E.153	—, —, —	—	—	—	—	3,765	—	—	—
Eland E.151	10, 6, 3, + Auxy.	36·0	104+	1,661+	—	—	12,500	—	—
Oryx	Ax, , + Auxy.	19·25	76·0	495	—	750	—	0·68	—
Rolls-Royce									
Derwent 8	C, 9, 1	43·0	83·1	1,280	3,600	—	14,700	1·01	—
Nene 10	C, 9, 1	49·5	104·0	1,610	5,100	—	12,500	1·00	—
Dart 506	C+C, 7, 2	37·9	95·1	1,030	365	1,400	14,500	0·83	
Dart 510	C+C, 7, 2	37·9	97·0	1,110	370	1,600	14,500	0·705	
R.B.109	—+—, Cann, —+—	32·0	115·0	—	1,000	4,020	—	—	—
Avon RA.7	12, 8, 2	42·2	110·0	2,460	7,500	—	7,800	0·92	
Avon 502	12, 8, 2	42·2	103·0	2,242	7,350		7,800	0·89	
Avon RA.21	Ax	42·2	110·0	2,520	8,000	—	—	0·925	
Avon RA.28	Ax, Ann, —	41·5	113·3	2,900	11,000	—	—	0·86	
Avon RA.29	Ax, Ann, —	41·5	113·3	2,790	10,500	—	—	0·86	
Conway	—+—, —, —+—	—	—	—	—	—	—	—	
Soar RSr. 1	Ax, Ann, —	15·75	62·75	267	1,860	—	—	1·26	—
Orenda									
Orenda 9	10, 6, 1	42·0	121·3	2,650	6,500	—	7,800	1·00	—
Orenda 14	10, 6, 2	—	—	2,900	7,500	—	—	0·91	—
PS.13 Super Orenda	Ax, Ann, —	—	—	4,000+	20,000	—	—	—	—

Notes: The legend '14, 11, 2' in column two implies a 14-stage axial compressor, 11 can-type combustors and a two-stage turbine; '—+—, Cann, —+—' implies an unknown number of l-p compressor stages, an unknown number of h-p compressor stages, a cannular combustion system and an unknown number of h-p and l-p turbine stages; 'Ax,' 'C' and 'Ann' are abbreviations for axial and centrifugal compressors and annular combustion chambers. The s.f.c. figures (specific fuel consumptions) are in lb/hr/lb thrust or (for turboprops) lb/hr/e.h.p. The data for the B.E.25, Orpheus, Gyron, R.B.109 and Super Orenda are not confirmed by the manufacturers, but have been unofficially published elsewhere.

The single-stage turbine has a Jessop G.18B disc and blades of Nimonic 80 secured with fir-tree roots. No difficulty was experienced in attaining the designed maximum thrust at 10,000 r.p.m.

Orenda. The successful performance of the Chinook, extensively used for research and development studies, confirmed the basic design of the larger Orenda which had been started late in 1946. Its main difference is the increase in the number of compressor stages from nine to ten. The six combustion chambers are relatively large, being approximately 12 in diameter and 24 in long.

First run in February 1949, in a little more than eight months it had completed more than 750 hr running with only routine maintenance. This time included six long runs under British, Canadian and American test schedules.

The first Orenda deliveries gave thrusts of rather less than 6,000 lb, the maximum rating then established. The Orenda 9, used in the first batches of CF-100 all-weather fighters, was rated at 6,500 lb, and later the figure was raised to 7,000 lb thrust. Present production engines are of the Series 11 and 14 types, and these have a two-stage turbine and a greater mass flow and pressure ratio. Over 2,000 Orendas had been delivered from Malton by the spring of 1955—a great achievement for a country which had no practical gas-turbine experience prior to 1947.

PS.13 Super Orenda. It has repeatedly been reported that Orenda Engines, Ltd., are developing a very large turbojet with this name (it was previously reported as being the ‘ Waconda ’). It is, according to unconfirmed reports, a large two-spool axial engine, incorporating a considerable proportion of titanium alloy, and with a thrust of between 15,000 and 20,000 lb. It is scheduled as the power unit of the twin-engined CF-105 delta-wing supersonic fighter.

CHAPTER 11

American Gas Turbines

POSSESSING such an abundance of research and production facilities backed by quite exceptional financial resources it is not surprising that America, despite a late start, is reducing the lead held by Britain in the design and development of aircraft turbines. That Rolls-Royce, Armstrong Siddeley and Bristol turbojets are being built under licence in America may be regarded as a tribute to British achievement and an indication of the quality of our engines.

America's active interest in the aircraft gas turbine may be considered to date from 1941, when the late General H. H. Arnold, C.-in-C. U.S. Army Air Forces, was so impressed by the performance of the turbojets he saw in flight in Britain that he arranged for a Whittle unit to be shipped to the U.S.A. This subsequently became the prototype of the General Electric units installed in the Bell XP-59A Airacomet experimental fighter. By April 1943, the General Electric I-16, based upon Whittle patents, was rated at 1,650 lb static thrust at 16,500 r.p.m., and a small production order was placed. In the following decade development has been rapid.

The companies principally concerned with the design and development of aircraft turbine power units are : Allison Division of General Motors Corporation ; Boeing Airplane Company ; Continental Motors Corporation ; Fairchild Engine and Airplane Corp. ; General Electric Company ; Pratt & Whitney Aircraft ; Westinghouse Electric Corporation ; and Wright Aeronautical Division of Curtiss-Wright Corporation.

ALLISON

J33-A-35. This is a late version of the I-40 turbojet, designed by General Electric and developed and produced by Allison. A variant (400-C4) was the first American turbojet to be approved for commercial operation. The double-sided impeller has 17 vanes on each side and the magnesium alloy diffuser has 14 tangential outlets and vaned elbows connecting with the combustion chambers. Supported in one ball bearing and one roller bearing, the impeller shaft is connected by a flexible splined coupling to the turbine rotor shaft. The compression ratio is 4·4 : 1 and the air mass flow 87 lb/sec at 11,750 r.p.m. at sea level.

Combustion chambers are of stainless steel and each houses a perforated stainless-steel flame tube and a fuel nozzle injecting downstream. The single-stage turbine has 48 inserted cast-steel guide vanes and the rotor disc is of forged stainless steel with 54 inserted solid blades. Cooling for the rotor is by air tapped from outside the nacelle.

Equipped with an afterburner, which increases the overall length to 213 in, the maximum output is raised to 7,000 lb. Alternatively, a water-methanol injection system can be used to augment the thrust. The J33 is approved for 600 hr running between overhauls, or up to 1,200 hr if operated from a base possessing facilities for minor repairs.

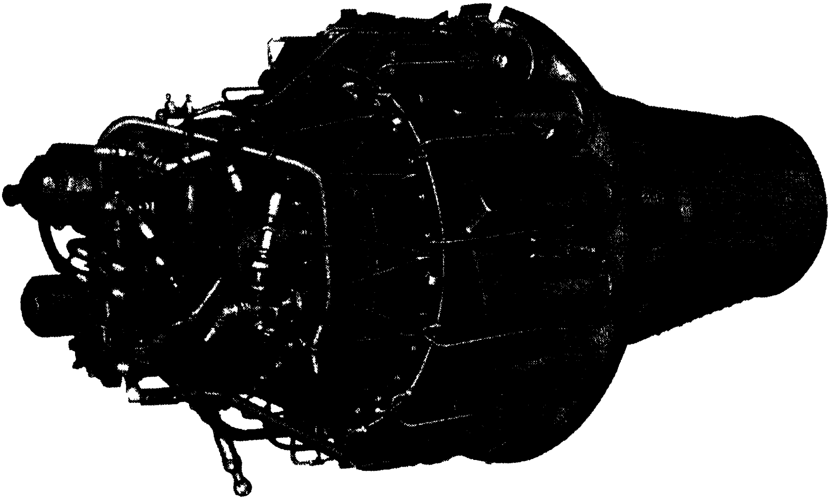


Fig. 11 : 1—Allison J33, 5,200 lb thrust.

Since 1950 a complete range of considerably modified variants of the Allison J33 have been developed for use in unmanned aircraft. These engines are simpler, lighter and generally more powerful than their predecessors. They run at higher r.p.m. and gas temperatures, and are generally ‘ lifed ’ at 10 hr. These new J33s are in large-scale production for such pilotless aircraft as the Martin TM-61A Matador and Chance Vought Regulus.

J35-A-33. Originally designated TG-180, this axial-flow turbojet was also of General Electric design. For the 11-stage compressor the light-alloy casing carries one row of inlet guide vanes, 11 rows of stator blades and one row of straightening blades. The rotor comprises ten light-alloy discs and one steel disc shrunk on a hollow shaft supported on a roller bearing and a ball thrust bearing. Both rotor and stator blades are of steel. Compression ratio is 5·1 : 1 and the air mass flow is 85 lb/sec.

Stainless steel is used for both casings and flame tubes of the eight combustion chambers into which fuel is sprayed by duplex-type burners.

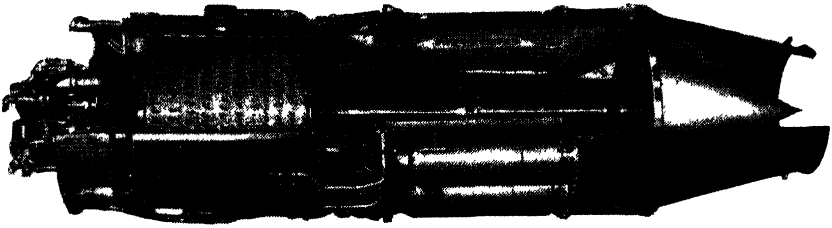


Fig. 11 : 2 Allison J35 with 11-stage axial compressor.

The single-stage turbine disc is welded to its shaft and makes a splined connection with the compressor rotor drawn up by an axial bolt. In the early versions the rotor had 126 blades but these have now been reduced to 95 in number.

An afterburner has been developed for the J35. This adds 375 lb weight and 49 5 in to the overall length but raises the thrust from 5,600 to more than 7,000 lb. Time between overhauls is already approved at 800 hr.

Approximately 12,000 Allison J35s have been built, principally for the Republic ' straight-wing ' F-84 and the Northrop F-89.

J71-A-1. Preserving the same modest diameter of 37 in, the latest Allison turbojet, the J71, develops a maximum thrust of 9,700 lb at 8,000 r.p.m. and, it is claimed, approximately 15,000 lb thrust with afterburning. Major features of the design are a 16-stage axial compressor and a three-stage turbine mounted on the same shaft. The combustion chamber is of the cannular type with ten flame tubes.

De-icing of the intake section is by air tapped from the main compressor and retractable screens at the intake prevent the ingress of foreign bodies when running on the ground. The compressor casing is built in

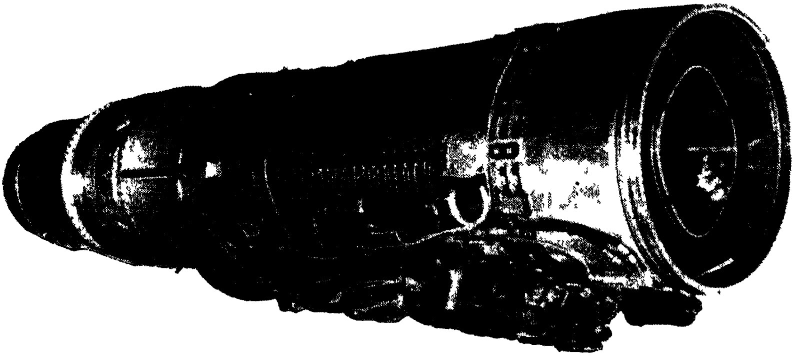


Fig. 11 : 3—Allison J71 turbojet has a 16-stage compressor.

two sections; the l-p portion is in magnesium and the h-p part in steel. Various types of J71 are used in the F-105, B-66 and F3H. Four afterburning J71s power the P6M SeaMaster flying boat.

T38-A-6. The Allison direct-connected turboprop was first flight-tested in a Boeing bomber in May 1949. It is a relatively high-speed unit, attaining its maximum output of 2,763 e.h.p. at 14,300 r.p.m. The compressor, having no less than 19 stages and giving a ratio of 6·5 : 1, is driven by a four-stage turbine. Following the technique used on Bell Aircraft war-time fighters, the airscrew reduction and driving gear is driven through extension shafting, from the nose of the compressor, to permit the most advantageous location of power unit and airscrew on the airframe.

T40-A-6A. Developed specially for the U.S. Navy, the T40 is a coupled version of the T38. As before, the drive is by extension shafts to the separate unit reduction gear for the counter-rotating airscrew. Each of the twin power units can operate independently, but each drives both airscrew elements. Cruising on one unit, therefore, does not entail the drag of a feathered airscrew. With an output of 5,525 e.h.p. and a specific weight of approximately 0·5 lb/e.h.p., four of these coupled units power the huge Convair Tradewind cargo flying boat.

In the Tradewind, an original scheme in which the T40 was mounted aft of the rear spar, driving the airscrews through long shafting, was later abandoned in favour of a more conventional installation. The produc-

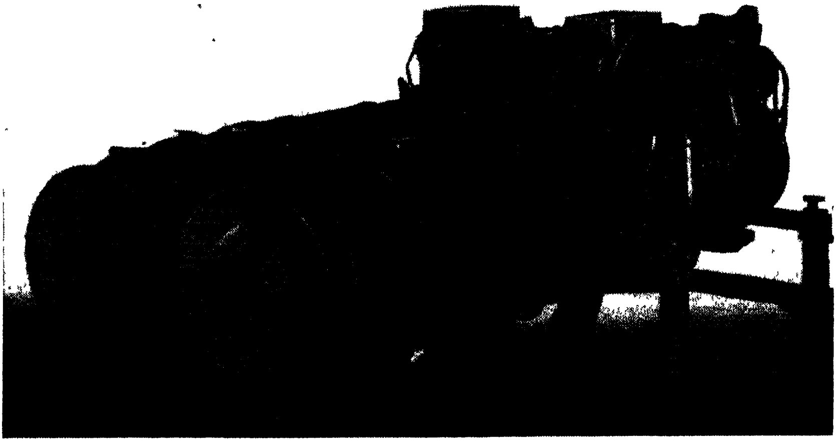


Fig. 11 : 4—Allison T40 coupled turboprop, 5,525 e.h.p., less drive shafts and gearbox.

tion machine has a pair of T40-A-10s mounted above the wing, with the engines drawing air from a plenum chamber fed from an intake beneath the spinner. This arrangement produces complete separation of water and ice from the intake air, the T40 being very susceptible to their ingress.

T54. Similar in most respects to the T40, the T54 consists of a coupled pair of T56 units. The engine is rated at about 7,500 e.h.p., and in the Republic XF-84H is geared to a supersonic airscrew.

T56. This single-shaft turboprop represents a very considerable advance upon the T38, although the two units are superficially not dissimilar. The T56 is an all-steel engine with a 14-stage compressor of which the design is noteworthy in achieving a 9 : 1 pressure ratio at sea level at the maximum rated power of 3,750 e.h.p. The cannular combustion chamber contains six flame tubes, and the turbine has four stages.

The engine speed of up to 13,820 r.p.m. is stepped down in a 12·5 : 1 ratio by the two-stage gearbox housed in a cast magnesium shell carried ahead of and above the centre-line of the engine. The T56 drives a single 13 ft 6 in three-blade airscrew. It is in production for the Lockheed C-130A Hercules military transport, and a civil version is designated Type 501. Development up to 4,500 e.h.p. is expected.

BOEING

As part of a long-term power plant research programme the Boeing Airplane Company has developed two miniature gas turbine units—the Model 500 turbojet and Model 502 (T50-BO-1) turboprop. Other Boeing undertakings include research on axial compressors and on ramjet units for the USAF.

Model 500. Weighing a mere 85 lb, this diminutive turbojet has a single-stage centrifugal compressor and a single-stage turbine. The compressor has a two-piece aluminium-alloy casing, with integral diffuser and two tangential outlets. The aluminium-alloy impeller comprises an inducer section with 28 blades, attached to the front of the main impeller, likewise with 28 blades. A compression ratio of 3 : 1 is attained and the air mass flow is 3·25 lb/sec at 36,000 r.p.m. There are two independent (not interconnected) tubular stainless-steel combustion chambers each with a perforated stainless-steel flame tube.

The turbine has a steel nozzle ring with 27 inserted guide vanes, and the rotor disc has 64 solid blades welded to the rim. At the maximum rating of 150 lb thrust at 36,000 r.p.m. the gas temperature at the turbine nozzle is 815° C.

The Model 500 turbojet is suitable for use as a power unit for missiles, glide bombs, or light aircraft, or as a starting unit for large turbines.

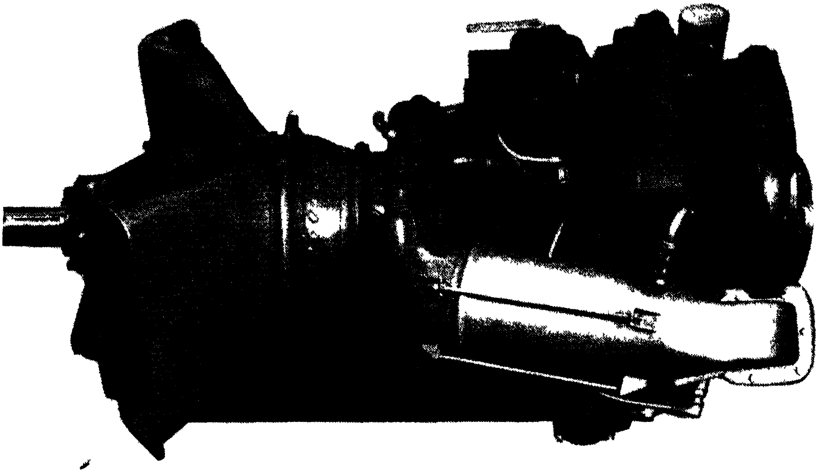


Fig. 11 : 5 Boeing T50-BO-1 turboprop.

T50-BO-1. This small turboprop is of the free-turbine type. The 7·5 in diameter h-p turbine is coupled to the compressor and the independent l-p turbine drives the airscrew through a 9·6 : 1 planetary reduction gear. Differing from most units of this type, the drive is taken from the rear or turbine end, leaving the forward intake to the compressor completely unrestricted. Maximum power for take-off is 210 e.h.p. with the compressor running at 37,500 r.p.m. and the power turbine at approximately 23,000 r.p.m. giving an airscrew shaft speed of 2,400 r.p.m. The compressor can be accelerated from an idling speed of about 10,000 r.p.m. to maximum in 5 sec.

In 1954 a more powerful and efficient version of the engine was produced, designated Boeing 502-10. Modifications have been made to the main rotating assembly enabling the unit to run at up to 270 h.p. at 3,100 output-shaft r.p.m. The rated power is 240 h.p. at 2,900 output r.p.m., an increase of 65 h.p. over previous models ; specific fuel consumption is also claimed to have been improved by 25 per cent.

A variety of applications is projected for the T50. It could be used to power light aircraft, helicopters, target planes, or boats, or serve as an auxiliary power unit for large aircraft. An experimental adaptation for a road vehicle power plant is described in Chapter 16.

CONTINENTAL

This company builds a range of Turboméca units under licence. Probably the most important is the J69 Marboré, which is being used to power a number of Service aircraft.

The original Marboré design has been developed through four principal versions, raising the thrust from 650 to 1,000 lb. The Continental J69 has been developed along lines very similar to those followed in France, although the American engines have rather more comprehensive accessory systems, larger oil capacity and other minor modifications. The J69-T-9 is in production for the Cessna T-37A twin-engined trainer and other types of J69 are being built for pilotless aircraft such as the Ryan Firebee.

The Turboméca Palouste air-compressor and Artouste shaft-drive engine are also being made in small numbers. Both have been flown in American aircraft. The T51 Artouste has powered fixed- and rotary-wing machines ; in August 1954, when installed in a Sikorsky XH-39, it was instrumental in setting a new world's helicopter speed record of 156 m.p.h.

FAIRCHILD

J44. Now in large-scale production for the U.S. Navy and U.S. Air Force, this short-life turbojet is remarkable for its plain cylindrical configuration. Measuring 22 in diameter and 72 in long, it develops approximately 1,000 lb thrust. Weight with accessory equipment is about 325 lb.

It presents a new constructional concept ; the conventional structure of rigid frames and housings is replaced by a sheet-metal cylinder connecting the two main bearing support frames and forming a pressure chamber enclosing the components. Castings and machined parts are reduced to a minimum, making it economical to produce and appropriate to expendable use in warfare. Mounting can be arranged by trunnions at the side of the compressor section and a stabilizing anchorage at the rear bearing frame, or by a cantilever attachment from the front of the compressor section and without any aft support. In certain installations the smooth outer shell makes the provision of additional cowling unnecessary. Being virtually self-contained it requires only the minimum fuel and control services from the airframe. Internal oil capacity is

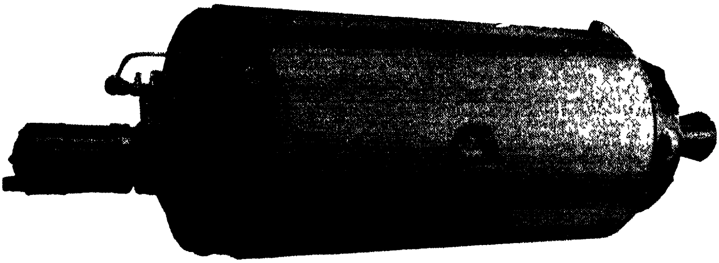


Fig. 11 : 6—Fairchild J44 ' monocoque ' turbojet, 1,000 lb thrust.

sufficient for 10 hr operation and there are no pumps, coolers or external oil lines. Controls, fuel pump and electrical units are mounted on a detachable accessory section. Only simple tools are required for maintenance or servicing. With the sole exception of repair or replacement of parts of the rotating assembly—which would necessitate access to balancing equipment—the power unit can be completely dismantled, overhauled and reassembled in the field.

The J44 has served as the power unit of the Ryan Firebee high-speed, high-altitude pilotless target plane. Obviously, it has other potentialities and is being used to power guided missiles. Mounted in small pods J44s would serve as auxiliary power boosters for existing aircraft.

The Fairchild J44 was also selected as the power unit for the first American vertical-rising jet-lift prototype, the Bell VTOL. In this aircraft a pair of J44s are mounted on pivots so that they can point vertically downwards for take-off or be swivelled to the normal horizontal position for forward flight.

GENERAL ELECTRIC

The first General Electric turbojet was the Model I-A, evolved from the Whittle W2B. Later came the I-14, I-16 and I-40, the last of these units being passed over for production to the Allison Division of General Motors. The first axial-flow unit of G.E. design was the TG-180, which is now produced by Allison as the J35. General Electric were further responsible for the design and development of the first American turboprop—the TG-100. The first engine mass-produced by G.E.C. was the TG-190, or J47.

J47. In its various type forms the General Electric J47 axial-flow turbojet is used to power a number of military aircraft. Basically, it has a 12-stage compressor driven by a single-stage turbine and giving an air mass flow of 90 lb/sec and a compression ratio of 5 : 1. Overall diameter is 36·75 in, length to tail cone 144 in and dry weight 2,500 lb. Maximum rating s.l.s. is 5,200 lb at 7,950 r.p.m. In normal operation it is rated for 1,200 hr between overhauls. Over 34,000 were on U.S.A.F. charge early in 1955.

J47-GE-17. A version for the North American F-86D all-weather fighter is equipped with an afterburner and variable nozzle which increases the length to 226 in and weight to 3,000 lb. The augmented thrust is reported to be approximately 8,000 lb. A completely automatic control system is fitted. It has been superseded by the J47-GE-33.

J47-GE-23. Originally intended for the Boeing B-47 six-jet bomber, this type was modified for ‘ all-weather ’ operation with hot-air de-icing

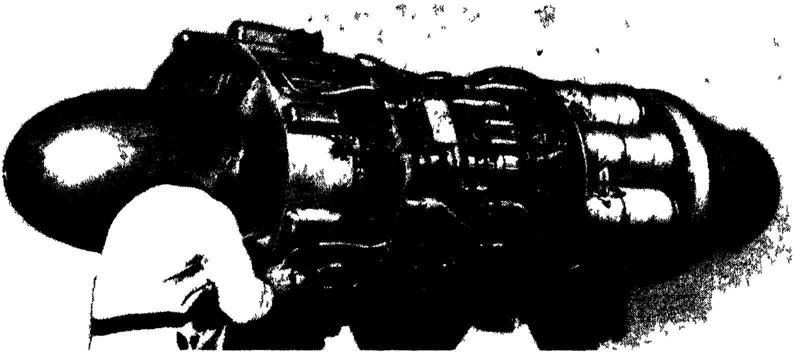


Fig. 11 7—General Electric J47-GE-23 all-weather turbojet, 5,800 lb thrust.

and high-altitude starting equipment. Maximum static rating was 5,800 lb thrust. By the end of 1954 approximately 15,000 J47s had been produced for the B-47 alone.

J73-GE-1 Retaining the same overall dimensions and characteristics, this unit is virtually a re-design of the J47, and during its early development period was designated J47-GE-21. The eight individual combustion chambers have been replaced by a cannular construction with ten

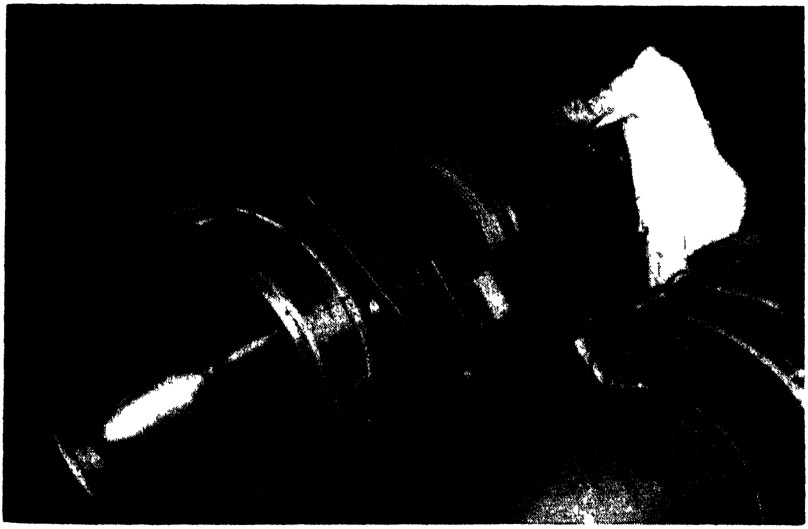


Fig. 11 : 8—General Electric J73-GE-1 turbojet with cannular combustion chamber. On the right is the nose of a J47-GE-17.

burners. A higher compression ratio is employed and this has led to the adoption of a two-stage turbine. Maximum output has been raised to 9,000 lb thrust and 14,000 lb is obtained with afterburning equipment. J73s are in limited production for the North American F-86H fighter-bomber.

J79. In the late 1940s the General Electric company built and ran a number of very large and powerful turbojets. One of these, the J53, recorded thrusts exceeding 17,000 lb, and reached 23,500 lb with an afterburner. None of these engines was placed in production.

The J79 is almost as large as the J53, and is designed to start its useful life with a rating exceeding 15,000 lb and to work up to about 25,000 lb by about 1960. It is intended specifically for supersonic applications, and relies to a large extent upon ram compression. The axial compressor is comparatively short and employs variable-pitch stator blading The whole structure of the engine is of steel.

The J79 is going into production for the supersonic Convair B-58A medium bomber, which has its J79s mounted in pods along a very thin delta wing. Another application of the J79 is in the later versions of the Lockheed F-104 fighter.

PRATT AND WHITNEY

A slightly modified version of the Rolls-Royce Nene, dubbed Turbo-Wasp, was produced in quantity by Pratt and Whitney. Typed as the J42 it achieved the distinction of approval for 1,000 hr running between overhauls.

J48-P-6. This designation refers to the latest model of the Turbo-Wasp, which is the American counterpart of the Rolls-Royce Tay. By a redesign of the rotating assembly—compressor impeller and turbine—the air mass flow of the Nene, 88 lb/sec, was raised to about 114 lb/sec for the Tay. With changes in the combustion equipment this enabled the static thrust to be increased from 5,000 lb to 6,250 lb. On the J48 this figure is raised to 7,000 lb with water-methanol injection or to

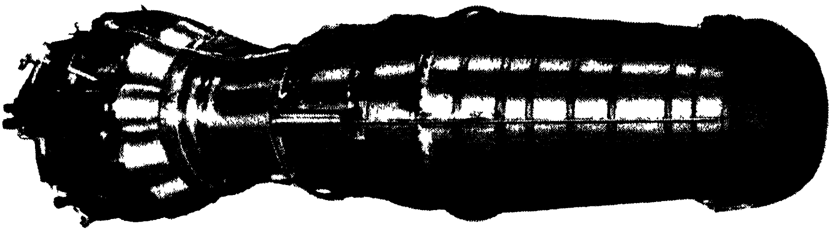


Fig. 11 : 9—Pratt and Whitney J48 turbojet with afterburner.

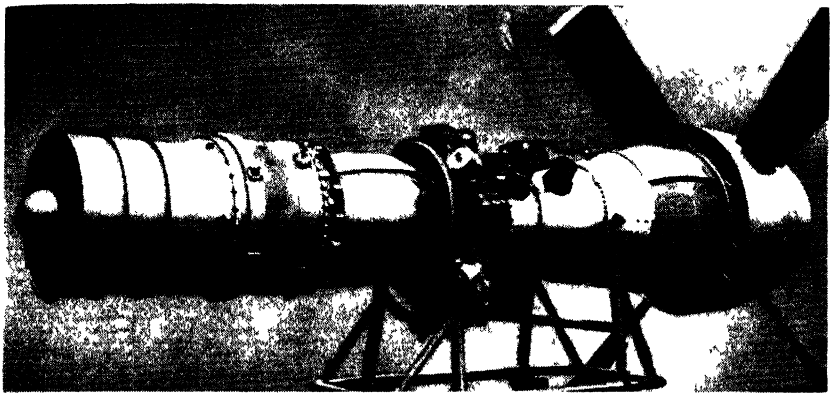


Fig. 11 : 10 – Pratt and Whitney T34 turboprop, 5,600 e.h.p.

8,750 lb when equipped with the afterburner, as illustrated. Large numbers have been built for Grumman naval fighters.

T34-P-2. Work on this high-power turboprop was commenced under U.S. Navy auspices in June 1945. In its developed form it has a 13-stage axial compressor, cannular combustion chamber and a three-stage turbine. Even amongst American units, which commonly make more use of stainless steel in construction than is usual in British practice, the T34 is remarkable as this metal is employed almost exclusively. Rated at 5,600 e.h.p. at 11,000 r.p.m. it is capable of 6,000 e.h.p. for take-off and weighs only 2,564 lb. Approximate overall dimensions are diameter 30 in and length 155 in.

The method of suspension is interesting. A heavy steel ring between compressor and combustion chamber carries four large rubber-bushed mountings which serve as pick-up points. No other support or steady point is provided. Four blow-off valves are fitted to the compressor, which is tapped for intake de-icing. The airscrew is driven through a two-stage gear giving a reduction of 11 : 1 and a brake is fitted to prevent windmilling when parked.

Various U.S.A.F. and U.S.N. developments of the Constellation, Stratocruiser and Globemaster transports have been fitted with the T34 A civil version, PT2F-1, was exhibited at the Paris Salon in 1953. For this the specific fuel consumption was 0·64 lb/e.h.p./hr at maximum power and 0·77 lb/e.h.p./hr at the sea-level cruising rating of 2,990 e.h.p.

J57. First delivered in February 1953, at which time it was already in large-scale production at both the Pratt and Whitney works at Hartford

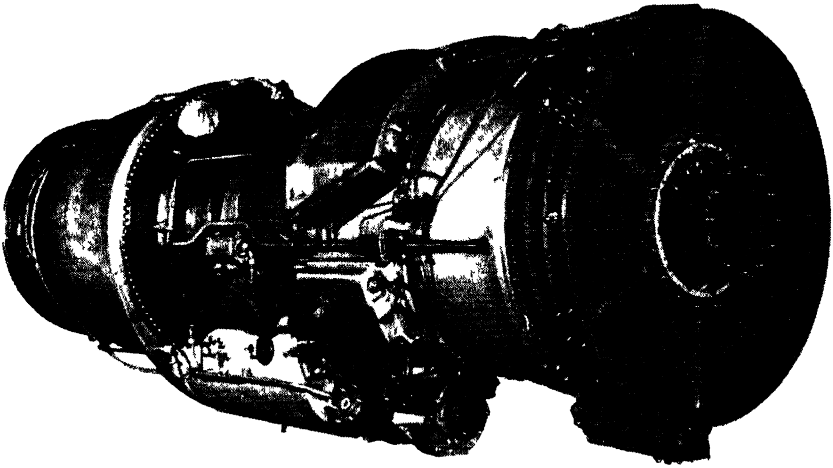


Fig. 11 : 11—Pratt and Whitney J57 two-spool-compressor unit. Rated output 10,000 lb thrust. This example is a bomber engine, without afterburner.

and the huge Ford plant at Chicago, this axial-flow, two-spool-compressor turbojet marks an important phase in the development of high-powered American units. The rated output is 10,000 lb thrust, which can be raised to 15,000 lb by operation of the Pratt and Whitney afterburner. It powered the North American F-100 Super Sabre in which the world speed record of 755 m.p.h. was established, and is the standard engine for numerous fighters, bombers, and both civil and military transports.

Structurally the unit is exceptionally rigid, being built up of nine drums bolted together at their peripheral flanges. None of these casings is longitudinally divided but, as steel is used throughout, the unit is relatively heavy at 4,400 lb. A nine-stage l-p compressor driven by a two-stage l-p turbine delivers air through a diffuser section to a seven-stage h-p compressor connected by a hollow drive shaft to a single-stage h-p turbine. The overall compression ratio is about 12 : 1 and the combustion chamber is of the cannular type. At full power the fuel consumption rate is 0·76 lb/hr/lb thrust. An electronic control system is employed to co-ordinate fuel flow rate, jet pipe temperature and rotational speed. It is operated from a single lever and, when required, it also controls the fuel flow to the afterburner and the area of the propulsion nozzle to maintain optimum performance.

Starting is effected by an air turbine mounted below the diffuser section and fed either from a service truck or, in the case of multi-engined aircraft, from the compressors of running units. The starting turbine drives only the h-p compressor and h-p turbine and the l-p rotating assembly is brought in by the gas flow after combustion has been initiated.

The main alternator is driven through a constant-speed hydraulic coupling from the forward end of the l-p compressor but other aircraft auxiliaries are remotely sited and driven by air tapped from the diffuser section.

More than half the total production of the J57 is of engines equipped with afterburners. Various types of short and long afterburner have been developed for such aircraft as the Super Sabre, Skyray and Voodoo. The maximum (reheat) thrust of these engines varies between 14,500 and 16,000 lb.

Pratt and Whitney are also developing turboprops based on the J57 design. These may be expected to have power outputs of from 10,000 to 12,000 e.h.p.

J75. Another unit, reported to have a designed thrust of 15,000 lb, is under development.

WESTINGHOUSE

One of the earliest American builders of turbojets, Westinghouse has concentrated from the beginning upon axial-flow types. Units from the smallest size have been produced experimentally, including the 9 5-B (Baby) intended to power a guided missile. This had a nominal diameter of 9·5 in and produced a thrust of 275 lb at 34,000 r.p.m.

The first Westinghouse turbojet to be placed in limited production was the 19-B (Yankee) which powered the McDonnell FD-1 Phantom fighter. Of relatively advanced design it had a six-stage compressor, 24-burner annular combustion chamber, and a single-stage turbine. A ' bullet ' control was provided to vary the area of the propulsion nozzle, as on the German Junkers jets. With an overall diameter of 20·75 in and weighing 847 lb it had an output of 1,365 lb thrust at a rotational speed of 18,000 r.p.m. Eventually this unit was developed as the J30, giving

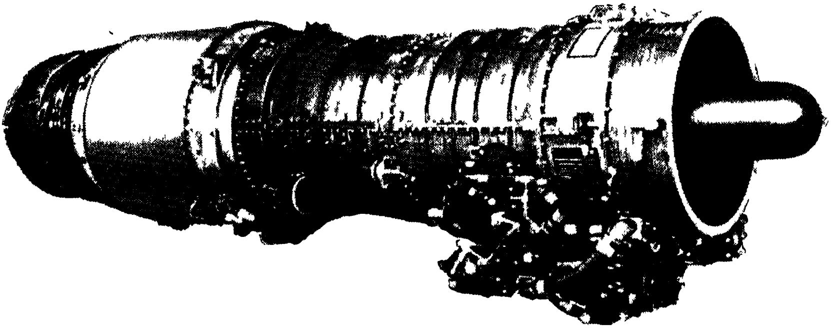


Fig. 11 : 12—Westinghouse J34 turbojet, 3,400 lb thrust.

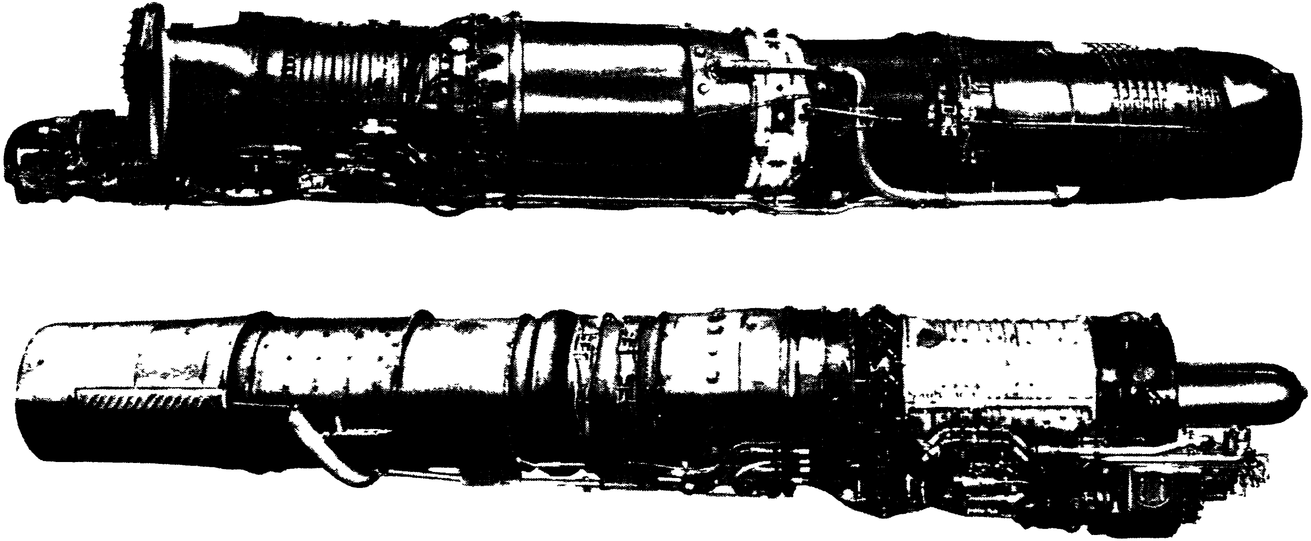


Fig. 11 : 13—Westinghouse J40 with afterburner for McDonnell Demon fighter (upper) and the J46 as scheduled for the Chance-Vought Cutlass (lower) Overall length in each instance is approximately 25 ft.

a maximum thrust of 1,600 lb at 17,000 r.p.m. and weighing only 692 lb.

A somewhat larger model, the 24-C with a 19-stage compressor and two-stage turbine, yielded 3,000 lb thrust and was the prototype of the J34 unit.

J34. Two versions of this type have been produced and the J34-WE-34 is approved for civil operation at a thrust rating of 3,250 lb at 12,500 r.p.m. Basic features are the 11-stage compressor with the low pressure ratio of 3·85 : 1, annular combustion chamber and two-stage turbine. Maximum gas temperature at the turbine nozzle is 775° C. Dry weight is 1,233 lb.

Maintaining the same diameter of only 24 in, the J34-WE-36 has the compression ratio increased to 4·35 : 1 and the gas temperature to 809° C to give a thrust of 3,400 lb at the same rotational speed. Length is reduced from 122 in to 111·4 in and the weight to 1,207 lb. The J34 has been used to power large numbers of Douglas Skyknight and McDonnell Banshee fighter aircraft.

J40. Originally scheduled for single-engined fighter installations, one version of this large turbojet has a bifurcated intake with the accessory equipment located between the arms of the duct. Main constructional features resemble the J34 but the compressor has only ten stages. Overall diameter is increased to 36 in and weight is approximately 3,000 lb. Maximum output s.l.s. of the J40-WE-6 is 7,500 lb thrust and specific fuel consumption is 0·95 lb/hr/lb.

A version equipped with afterburner and designated J40-WE-8 is used to power the McDonnell F3H-1 Demon fighter. It has been reported to deliver a maximum thrust of 11,600 lb. The overall length of this unit is about 25 ft.

J46. This unit follows the general design of the J34 and was selected to power the Chance-Vought Cutlass and the Convair Sea Dart. Its output is in excess of 4,800 lb thrust and an afterburner with a variable-area nozzle is available (afterburning thrust is over 6,000 lb).

In 1953 Westinghouse entered into an agreement with Rolls-Royce for technical co-operation, interchange of information and, by arrangement, manufacture of developed units under licence. Fruitful results may be expected from such a pooling of resources and experience.

WRIGHT

Initially the Wright Corporation took over the development of the Lockheed L-4000 axial-flow turbojet and, as XJ-37, raised the static thrust to 5,500 lb. An axial-flow turboprop, XT-35, producing about 5,500 e.h.p. was also built, tested and flown.

AMERICAN AIRCRAFT TURBINES

Manufacture and Type	*Compressor Stage and Type*	*Combustion Chambers*	*Turbine Stages*	*Max. Power Rating (Sea level static)*			*Diameter in*	*Length in*	*Weight lb*
				Thrust lb	*Equivalent h.p.*	*Speed r.p.m.*			
Allison									
J33-A-35	1 C	14	1	5,200	—	11,750	50·5	107·0	1,820
J35-A-33	11 A	8	1	5,600	—	7,800	37·0	146·0	2,305
J71-A-1	16 A	Cann	3	9,700	—	8,000	37·0	179·0	3,650
T38-A-6	19 A	8	4	—	2,763	14,300	—	149·0	1,540
T40-A-6A	19 A	8	4	—	5,525	14,300		185·0	2,575
T56-A-1	14 A	Cann	4	—	3,750	13,820	27·0	145·0	1,610
Boeing									
Model 500	1 C	2	1	150	—	36,000	22·0	29·0	85
Model 502	1 C	2	2	—	210	37,500	23·3	48·1	267
Fairchild									
J44	1 C	Cann	1	1,000	—	15,750	22·0	72·0	325
General Electric									
J47-GE-13	12 A	8	1	5,200	—	7,950	36·75	144·0	2,500
J47-GE-23	12 A	8	1	5,800	—	7,950	39·5	144·0	2,650
J73-GE-1	12 A	Cann	2	9,000	—	8,000	39·5	146·0	3,400
Pratt & Whitney									
J48-P-6	1 C	9	1	6,250	—		50·0	106·75	2,000
T34-P-2	13 A	Cann	3	—	5,600	11,000	34·0	155·0	2,564
J57	9 A 7 A	Cann	1 (h.p. comp.) 2 (l.p.) (comp.)	10,000	—		41	141	4,400
Westinghouse									
J34-WE-34	11 A	Ann	2	3,250	—	12,500	24·0	122·0	1,233
J34-WE-36	11 A	Ann	2	3,400	-	12,500	24·0	111·4	1,207
J40-WE-6	10 A	Ann	2	7,500	—	7,600	40·0	192·0	3,000
J46-WE-8	11 A	Ann	2	4,800	—		36·0	198·0	2,100

Currently two British units are being built under licence agreements. With the U.S. designation J65 the Armstrong Siddeley Sapphire is already the standard power unit of the Martin B-57 (Canberra), the Republic F-84F and RF-84F, the North American FJ-3 and -4 Fury fighters, the Douglas A4D Skyhawk attack bomber and (with an afterburner) the Lockheed F-104 and Grumman F11F Tiger supersonic fighters.

The Wright Corporation is also building two turboprop units, T47 and T49, credited with outputs of 9,000 and 11,000 s.h.p. They are developments of the J65 and J67 driving large diameter Curtiss airscrews.

The Bristol Olympus two-spool turbojet will be known as the J67. It has undergone considerable modification and is now claimed to give a thrust of 15,000 lb on test. It will be fitted in the Republic F-103 and other fighters.

CHAPTER 12

European Gas Turbines

DESPITE the achievement, as a private venture of the Heinkel Co., of the world's first turbojet flight in 1939 and the decision of the German Air Ministry in the same year to support the development of turbine power units, production and utilization was delayed until the later stages of the war period. This would appear to be due more to lack of a firm, consistent and continuous policy, to the dispersion of effort and resources over a multitudinous range of projects, and to business rivalry than to any incapacity to solve the problems of development. Towards the end of the war, however, turbojets became very attractive to the Germans by reason of the reduction in materials and man-hours required for production as compared with those for a reciprocating engine and airscrew—this apart from the advantages of higher power output and lower specific weight offered by turbojets for the advanced aircraft under development. At the time of the collapse, the Junkers Jumo 004 turbojet, for example, had accumulated more than 25,000 hr of bench testing, over 5,000 units had been built, production was at the rate of 1,500 per month and was scheduled for eventual increase to 3,000 per month.

After an appraisal of German aircraft and turbojets the Royal Aircraft Establishment, Farnborough, reported in November 1945 :

> ' They (the Germans) tried desperately to stem the tide with jet fighters. To some extent these were successful, but jet propulsion did not come into use soon enough to have more than an infinitesimal effect on the war. If the end had not come when it did, it might have been otherwise. There is little doubt that in 1946 there would have been a severe struggle for fighter supremacy, the advantage in numbers being with the Germans, and in quality with us.'

Although in durability and efficiency those German turbojets that were operational at the end of the war were generally inferior to corresponding Rolls-Royce and de Havilland units, the quality of German airframe design, as manifested in the Me 262 fighter, promised to compensate in some degree for technical deficiencies in the power plants. It must be remembered, moreover, that Germany had amassed considerable operational experience with jet fighters and light jet bombers, and that the performance of new aircraft would have benefited to a very important extent from the rearward sweeping of wing and tail surfaces, to permit attainment of higher critical Mach Numbers.

Although Daimler-Benz and Focke-Wulf had designed units of their own, German gas turbine development was principally in the hands of B.M.W., Heinkel (at first alone, and later in collaboration with Hirth), and Junkers. Heinkel HeS 3B and HeS 8 units were completely outmoded by the HeS 011, work on which was begun in 1941 and which had reached a fairly advanced stage of development when hostilities ended. Junkers did some preliminary work on gas turbines as early as 1936, but little progress was made until late in 1939, when the design of the Jumo 004 was put in hand. After the war, it was revealed that the first unit of this type was tested about a year later and underwent flight tests in 1941. This early model gave a thrust of about 1,760 lb. The Jumo 004B, the sub-type selected for quantity production, was designed at the end of 1941, and was first run about 12 months later. A prototype Me 262 with two 004B units was flown early in 1943.

The B.M.W. concern, although engaged in jet research during 1937, began work on their 003 project in 1939, the designed static thrust being 1,300 lb. The first unit was tested in 1940, but delivered only 990 lb thrust, with a specific consumption of 2·2 lb/hr/lb.

Whereas all the early production-type turbojets produced in Great Britain, and later in America, had centrifugal compressors, all the German designers decided—quite independently, it would appear—to employ axial-flow compressors. This fact is interesting in view of Germany's wide experience with centrifugal compressors for various purposes. The designers realized the paramount necessity for ensuring the smallest possible diameter and thereby low drag, and were confident that the axial compressor, despite its current liability to stall, would ultimately prove to be the more efficient.

It is obvious from an examination of Germany's gas turbines that design was adversely affected by the need to conserve vital alloys. One of the outstanding efforts to overcome this handicap was the production of hollow turbine blades with internal passages for cooling air, fabricated from austenitic steel sheet. There was, of course, some loss in performance through diverting air from the compressor to cool the turbine in this way. Variable-area jet nozzles controlled by a 'bullet,' the use of a light diesel oil as fuel, and twin-cylinder two-stroke engines for starting, were features common to the three main types of German gas turbines.

B.M.W.

B.M.W. 003. This unit was smaller and lighter than the Jumo 004 and the thrust rather less at 1,760 lb. In layout it differed from the Junkers chiefly in having a single annular combustion chamber with 16 burners, a feature which gave an overall diameter of 27·1 in against 30·4 in for

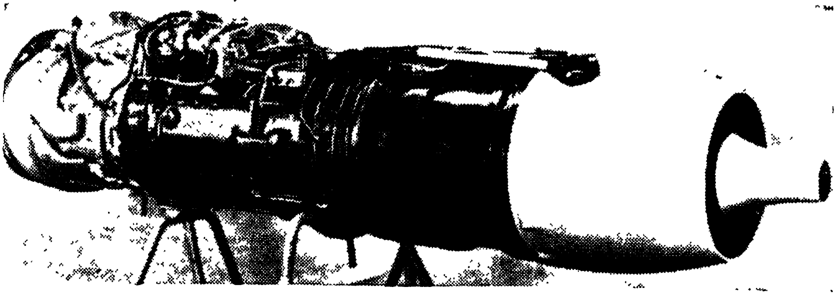


Fig. 12 : 1—B.M.W. 003 turbojet, 1,760 lb thrust.

the Jumo. The single-stage turbine had hollow air-cooled stator and rotor blades and the adjustable ‘ bullet ’ was internally cooled. Early experimental 003 units had six compressor stages but seven stages were standardized for production units and a new series (003D) was under development with eight stages.

Figures from the B.M.W. works show that air to cool the hollow stator and turbine blades consumed about 5 per cent of the compressor capacity. The compressor efficiency was said to be 78 per cent and that of the turbine 79 per cent. A life of 50 hr between overhauls was claimed, after which the turbine required careful inspection. The turbine rotor could be withdrawn and replaced in 2 hr without removing the complete unit from the aircraft. The total life of the combustion chamber was quoted as 200 hr. Dry weight of the complete unit was 1,344 lb.

Other B.M.W. projects under development included a turbojet with a 12-stage compressor, annular combustion chamber and three-stage turbine designed to produce 7,520 lb thrust at 6,000 r.p.m. A turboprop version of this unit, having a four-stage turbine, was expected to deliver 7,000 h.p. through counter-rotating airscrews.

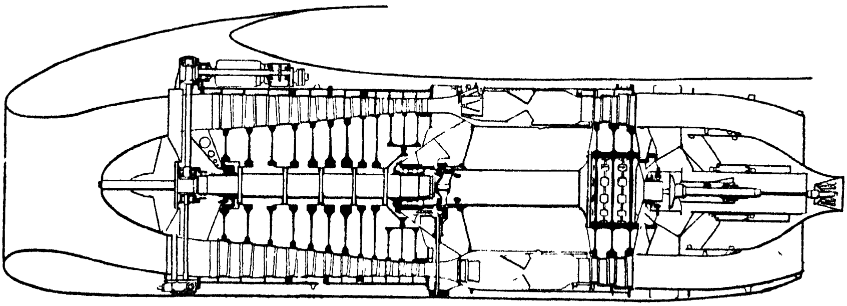


Fig. 12 : 2—B.M.W. 018 turbojet, estimated 7,520 lb thrust.

DAIMLER-BENZ

D-B 007. The designation ZTL was applied to the Daimler-Benz 007 turbojet developed by Professor Leist. It indicated Zweikreis-Turbine-Luftstrahl—literally two-circuit turbojet or turbojet with augmenter. It was tested in the autumn of 1943 but work was stopped by the German Air Ministry on the grounds of its over-complexity. Nevertheless, the unit was of great technical interest. A compressor and ducted fan were mounted on two counter-rotating drums, the inner directly driven by the turbine and the outer through gears at about half speed. The inner drum carried nine stages of compressor blading while the outer drum had eight stages of compressor blading internally and three stages of fan blading externally. The turbine rotor was cooled over 30 per cent of its circumference by air drawn from the ducted fan circuit, the remaining 70 per cent of the circumference receiving the working gases. Four tubular combustion chambers were used, and provision was made for a fifth in case it was found possible to reduce the amount of turbine cooling air.

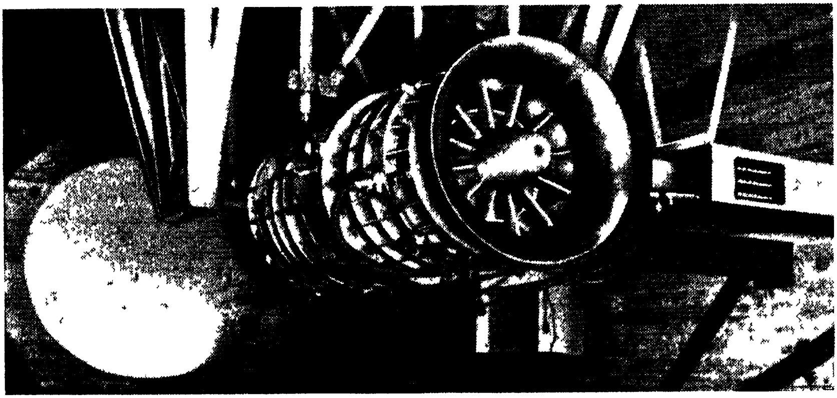


Fig. 12 · 3 Daimler-Benz turbojet with counter-rotating compressor and ducted fan

The unit, which weighed 2,870 lb, was 182 in long, and 33.5 in diameter. Under the designed conditions of operation, 550 m.p.h. at 19,000 ft, the thrust was calculated as 1,363 lb. Other data are: air mass flow through compressor, 17.6 lb/sec; air mass flow through fan, 35.2 lb/sec; fuel consumption, 0.55 lb/hr/lb; speed of turbine and inner compressor rotor, 12,000 r.p.m.

HEINKEL-HIRTH

HeS 011. Designed by Herr Ohain, the Heinkel-Hirth HeS 011 was of interest in that the compressor had three axial stages preceded by a

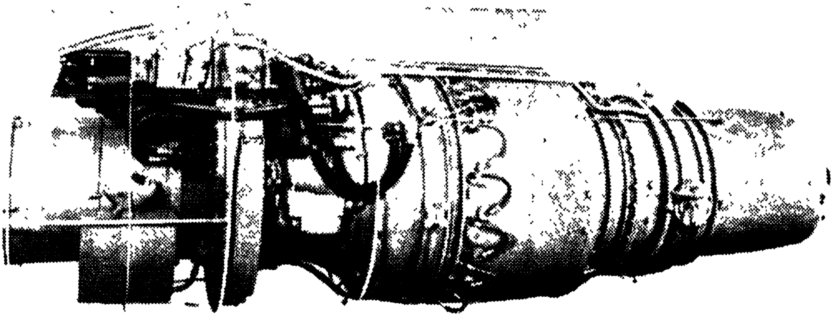


Fig. 12 : 4—Heinkel-Hirth HeS 011 turbojet.

'diagonal' stage which was partly axial and partly radial. An axial-flow inducer at the intake was another novel feature. The combustion chamber was of the annular type furnished with 16 fuel sprayers. Hollow blades were fitted to the two-stage turbine rotor. The adjustable 'bullet' in the tail pipe had two positions only: fully 'in' for idling and fully 'out' for all other conditions. Thrust was 2,860 lb, rotational speed 10,000 r.p.m., and weight 2,090 lb.

JUNKERS

Jumo 004. The most famous of the German war-time turbojets, this unit had an eight-stage axial compressor and a large-diameter single-stage turbine. The compressor casing, divided on an axial plane, was of cast magnesium, and stator blades were assembled in half rings and bolted into each casing half. The duralumin blades of the compressor rotor were dovetailed into staggered grooves on the periphery of light-alloy discs and were fixed by grub screws through each root.

Entry guide vanes and the first row of stator blades were of fairly

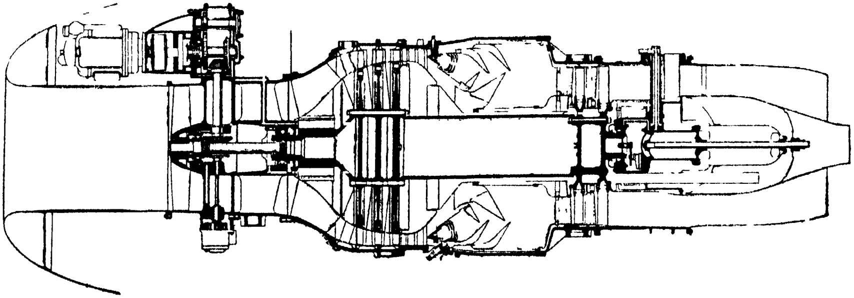
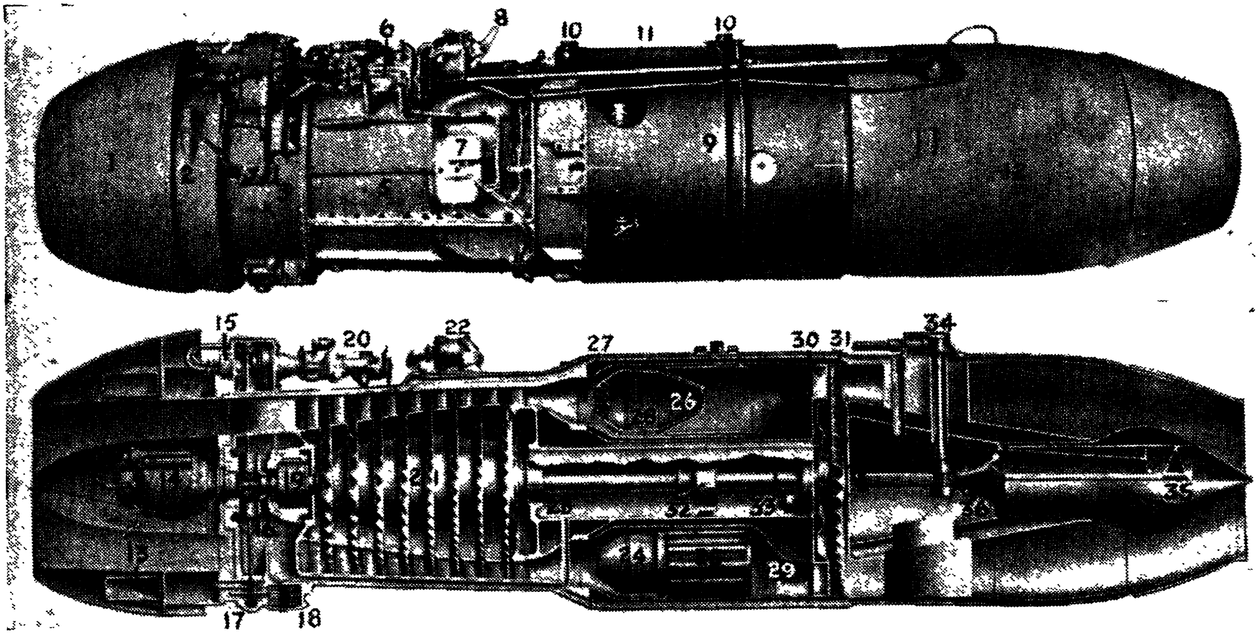


Fig. 12 : 5—The HeS 011 compressor had an axial inducer, a 'diagonal' stage and three axial stages.



1. Nose cowling: 2. Oil tank: 3. Entry casing: 4. Auxiliary gear box: 5. Compressor casing· 6. Servo motor: 7. Ignition apparatus: 8. Control lever: 9. Outer casting: 10. Attachment points: 11. Bullet control shaft: 12. Exhaust casing: 13. Annular fuel tank. 14. Riedel starter: 15. Injection pump: 16 Auxiliary drive: 17. Oil pump: 18. Oil filter: 19. Front compressor bearing: 20. Speed regulator: 21. Compressor rotor: 22. Fuel filter: 23. Rear compressor bearing: 24. Flame tube: 25. Combustion chamber muffle: 26. Diffusing grille: 27. Combustion chamber: 28. Injection nozzle: 29. Turbine entry casing: 30. Turbine stator blades: 31. Turbine: 32. Forward turbine bearing: 33. Rear turbine bearing with oil scavenging pump: 34. Bullet-operating gears: 35. Movable bullet: 36. Bullet support.

Fig. 12 : 6—Junkers Jumo 004 unit: port side view and longitudinal section.

thick aerofoil-section light alloy, the second stator row being of thinner section, and the remainder of cambered sheet steel. The rotor was built up with two steel shafts attached to the outside faces of the first and last discs. At the front the compressor bearing comprised three ball races, each capable of supporting end thrust, while the rear bearing was of a single roller race.

Cooling air was bled off between the fourth and fifth compressor stages and led into the double skin surrounding the combustion chamber assembly. A small amount of air was allowed to pass into the space between the combustion chambers and the inner wall. Most of the air passed down a strut to circulate inside the ' bullet ' and discharge through small holes to cool the downstream face of the turbine disc. Some of this cooling air also passed into a double skin which extended to within about two feet of the final nozzle. After the last compressor stage, air was bled off internally and taken through tunnels in two of the casing ribs to cool the upstream face of the turbine disc. More air was taken through three tunnels in the central casting into the space between the two plate diaphragms in front of the turbine ; most of this air then passed into the hollow turbine nozzle guide vanes, emerging through slits in the trailing edges of the vanes.

Efficiencies of 85 per cent and 79 per cent were claimed for compressor and turbine respectively. To cool the turbine blades and guide vanes, 7 per cent of the compressor delivery was diverted.

Six interconnected combustion chambers of aluminized steel were disposed parallel to, and evenly spaced around, the central casing carrying the rear compressor bearing and the turbine shaft bearing. A fuel nozzle in each chamber injected fuel upstream and igniter plugs were mounted in alternate chambers. Swirl vanes were fitted to the forward end of each chamber, with baffles at the rear, the hot gases passing through ' slot mixers ' formed in the rear conical wall. The hot gases then mixed with the air which by-passed the flame tubes.

There were 61 sheet steel blades on the turbine rotor. Hollow air-cooled blades were adopted only because of the time factor in evolving suitable material to withstand the extremely high temperatures, and the necessity of conserving strategic alloying metals. These blades had forged box-section roots and were fitted over lugs formed on the periphery of the disc and secured by 5 mm pins and a special soldering process.

Mounted in the tail pipe was a movable ' bullet ' operated by a servo-motor controlled from the throttle lever. A rack-and-pinion device moved it longitudinally. On the ground, or idling in the air, the ' bullet ' was held fully forward while the turbine was running up to 30 per cent of the maximum r.p.m. Between 30 per cent and 90 per cent of maximum r.p.m. it was moved rearwards to reduce the area of the jet orifice.

At take-off it was near the end of its backward travel. For maximum performance in flight, above 20,000 ft and at 400 m.p.h., it was moved to the extreme rearward position to provide maximum thrust. The servo-motor control was interlinked with a capsule sensitive to ram pressure so that the position of the ' bullet ' was adjusted according to the ram pressure and consequently to the forward speed of the aircraft.

A Riedel flat-twin, two-stroke petrol engine for starting was mounted in the air intake co-axially with the compressor shaft. From the cock-pit it could be started electrically, but on the ground manual starting by means of a cable and pulley was possible.

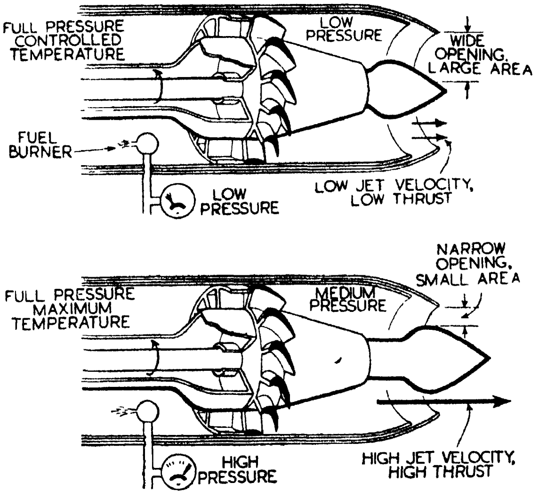

FULL PRESSURE CONTROLLED TEMPERATURE
LOW PRESSURE
WIDE OPENING, LARGE AREA
FUEL BURNER
LOW PRESSURE
LOW JET VELOCITY, LOW THRUST
NARROW OPENING, SMALL AREA
FULL PRESSURE MAXIMUM TEMPERATURE
MEDIUM PRESSURE
HIGH PRESSURE
HIGH JET VELOCITY, HIGH THRUST

Fig. 12 : 7—Diagram showing influence of ' bullet ' control on Jumo 004.

It was asserted that a Jumo 004 unit could be assembled in 50 man-hr and a complete overhaul and reassembly in 100 man-hr. In service a complete overhaul was undertaken after 30 hr operation.

FRANCE

DASSAULT

Although hitherto concerned solely with the design and construction of aircraft, this firm in 1953 acquired a licence to build the Armstrong Siddeley Viper turbojet. Under the designation of M.D.30, and probably with an afterburner, this engine is scheduled for production.

HISPANO

Building Rolls-Royce centrifugal-type turbojets under licence, this company has played an important part in the re-equipment of the French

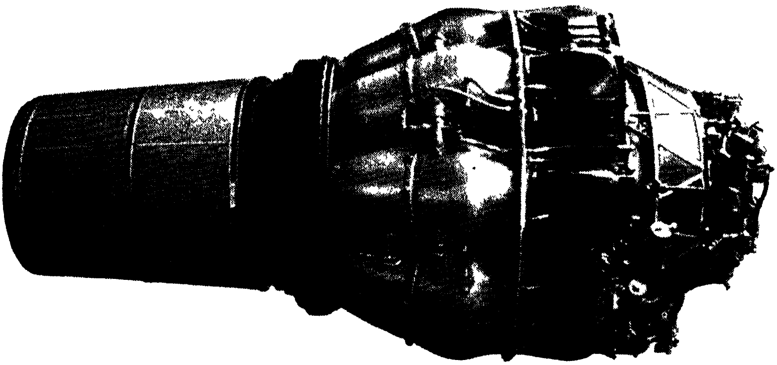


Fig. 12 : 8—Hispano Tay 253 develops 7,700 lb thrust or 9,920 lb with afterburner.

Air Force. The Hispano Nene has powered a number of fighter aircraft, including the Breguet Vultur, Dassault Ouragon, S.E.53 Mistral (Vampire), S.E.2410 Grognard and S.O 6025. Mostly these were of the Nene 102 type rated at 5,000 lb thrust but a later version, Nene 105, develops a maximum static thrust of 5,200 lb. Over 1,000 Hispano Nenes have been delivered.

While official policy terminated work on the Rolls-Royce Tay in Britain, Hispano actively continued development and the Tay 253 delivers 7,700 lb thrust. Equipped with an afterburner, which raises the thrust to 9,920 lb, it is designated R.450 Verdon and is installed in the Dassault Mystère IV.

Hispano has extended the licence to include the Rolls-Royce Avon

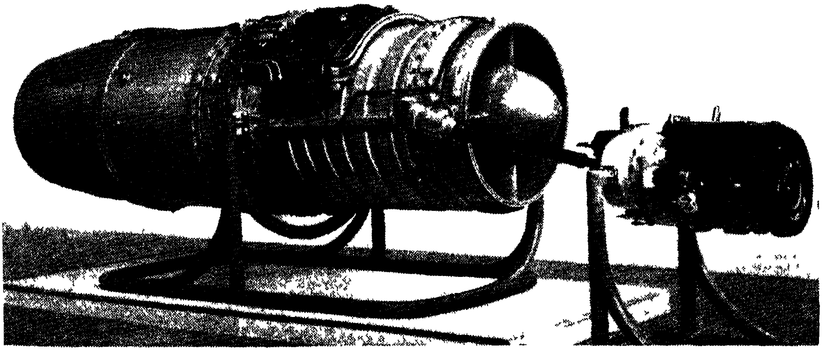


Fig. 12 : 9—S.N.E.C.M.A. Atar 101-B turbojet with separate unit starter and auxiliaries gearbox.

and will build an afterburning version of the Avon for the Mystère IV B.

S. N. E. C. M. A.

101-C Atar. The Voisin division of the S.N.E.C.M.A. organization in 1947 commenced the design of a 5,000 lb thrust turbojet based on the BMW 003 unit. It has been progressively developed through several type forms and the 101-C version is in production for the Mystère IV and the S.O. Vatour all-weather fighter.

Basically, the unit is of the axial-flow type with a seven-stage compressor, annular combustion chamber and a single-stage turbine. Compression ratio is about 4·2 : 1. For de-icing, air tapped from the main compressor is passed through the entry guide vanes, around the intake duct and through its support webs. Fuel is delivered to the combustion chamber by way of 20 h-p atomizers. Primary air is fed past the burners while secondary air jets, from concentric rings of slots located a few inches beyond the burners, continue the combustion and mixing process. The turbine rotor has 53 blades, drilled for air cooling and attached by Laval-type roots. Air cooling is also employed for the turbine nozzle. The 29 hollow guide vanes are of sheet metal and fitted with an insert member to direct the air stream over the inner surfaces. Early units—101-A and 101-B versions—had a variable-area propulsion nozzle of the internal ' bullet ' type but this has been superseded by a pair of eyelid shutters.

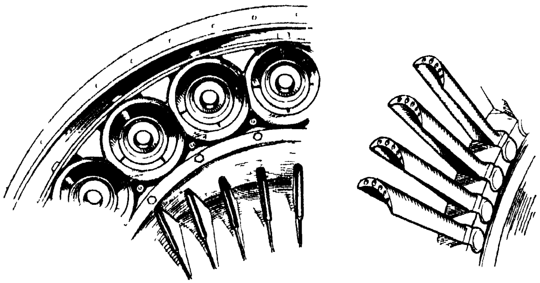


Fig. 12 : 10—Details of Atar 101-B fuel burner arrangement and of the air-cooled turbine blades.

A feature of the Atar is the provision of a remotely-sited starter motor and auxiliaries gearbox coupled by a universal-jointed shaft. The starter is a 25 h.p. two-stroke petrol engine of the firm's manufacture. Installation is simplified as the designer has wide scope in the location of auxiliaries. Operationally it is advantageous as, prior to starting the main unit, the engine can be run to drive the generator, compressor, vacuum or hydraulic pumps for essential services.

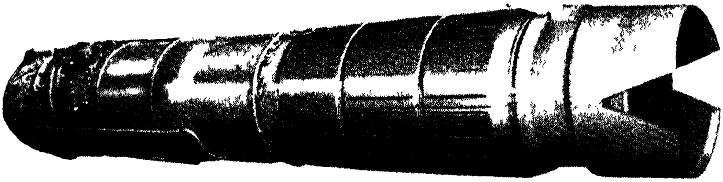


Fig. 12 : 11—Atar 101-F with afterburner delivers a total thrust of 8,370 lb.

Atar 101-C is rated at 6,170 lb thrust at 8,500 r.p m. Type 101-D delivers 6,610 lb thrust and, fitted with an afterburner and termed 101-F, a total of 8,370 lb. A later version, 101-G, with afterburning is in production for Mystères at 8,820 lb thrust.

Vulcain. This is a more powerful unit following the same general design with a seven-stage compressor and an annular combustion chamber with 20 burners. To date it has not been in the air but on the test bed has already been run at a thrust of 13,000 lb. It is to power several advanced designs of military aircraft. The basic weight is 3,080 lb.

TURBOMÉCA

A comprehensive range of small multi-purpose turbine units is produced by the Société Turboméca under Szydlowski patents. The units fall into four main type groups—turbojets, air compressors, single-shaft and two-shaft turbine units, and a by-pass jet—and a not inconsiderable standardization of components has been achieved.

Typical of the turbojets is the Palas (Fig. 12 : 13), which has a maximum thrust of 350 lb and weighs 159 lb. A centrifugal compressor feeds an air mass flow of 6·85 lb/sec at a pressure ratio of 3·9 : 1 to an annular

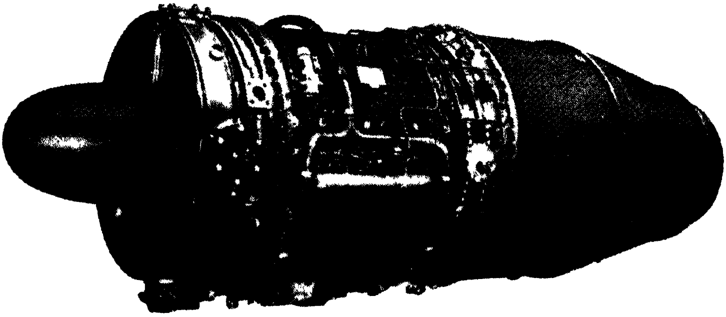


Fig. 12 : 12—S.N.E.C.M.A. Vulcain, expected to have a maximum rating of 13,000 lb.

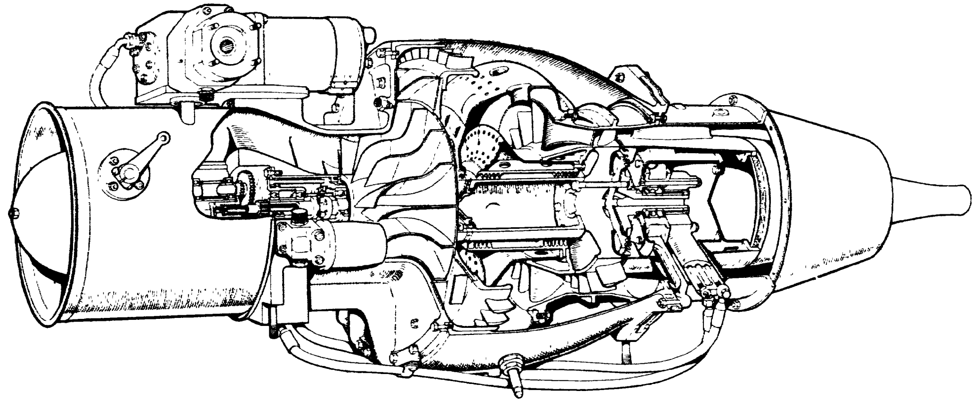


Fig. 12 : 13—Turboméca Palas turbojet has a maximum thrust of 350 lb.

combustion chamber. Fuel is sprayed radially into the chamber by a thrower disc, on the 'centrifugal injection' system described in Chapter 7. The fabricated nozzle guide vane assembly is of interest. From the outer diameter of the combustion chamber casing air passes through the hollow sheet-metal blades to reach the inner diameter of the casing and thence, by holes in the rear radial wall of the flame section, the primary combustion zone. Thus the blades are cooled and some part of the combustion air is pre-heated. Turbine blades are solid and integral with the disc

The main rotating assembly is supported on a ball bearing forward of the compressor impeller and a roller bearing aft of the turbine disc

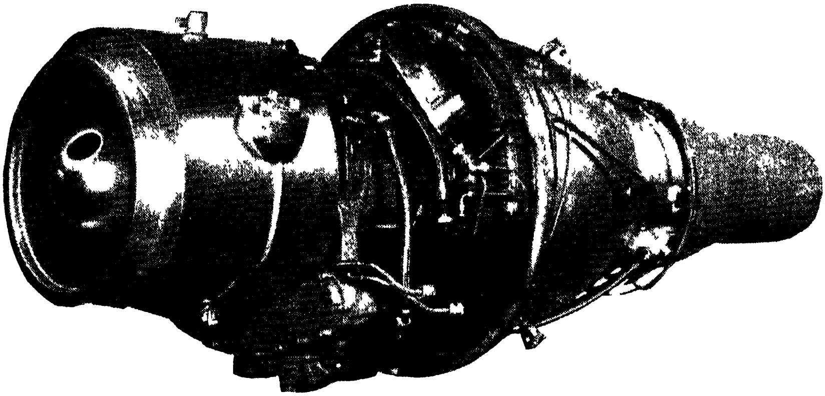


Fig. 12 14 Designated Continental J69 T-9, the Marboré is in production in the U.S.

To ensure alignment of bearings under the varying temperature conditions obtaining during operation, the rear bearing housing is not rigidly attached to the enveloping structure. Instead, it is mounted on three radial arms, pivoted at their outer ends to the terminal flange of the combustion chamber casing and at their inner ends to the bearing housing, passing through streamlined hollow struts spanning the exhaust duct.

Overall dimensions of the Palas are : diameter 15·95 in and length 47·2 in. Other jet units are the Piméné, thrust 220 lb, weight 121 5 lb, and Marboré, thrust 883 lb, weight 309 lb. They are suitable for powering target aircraft, guided missiles, jet trainers, light aircraft and sailplanes, or for auxiliary power boosters for piston-engined aircraft.

Three single-shaft turbine units for driving aviation or industrial auxiliary equipment resemble the jet units in general design but include a two-stage turbine rotor and a spur-type double-reduction gear. The gear and the output shaft are mounted forward of the compressor and carry the auxiliary equipment and drive. Designated Oredon, Artouste I and Artouste II they deliver 158, 276, and 394 b.h.p. and weigh 179, 183, and 200 lb respectively. Alternative reduction gear arrangements furnish maximum output speeds of 12,000 or 6,000 r.p.m.

By the addition of a secondary gearbox, reducing the shaft speed to 2,500 r.p m. for an airscrew, the Artouste II becomes a main propulsion unit delivering about 400 h.p. In this guise, termed the Marcadau, it will be installed for the forward propulsion of the later versions of the Farfadet convertiplane.

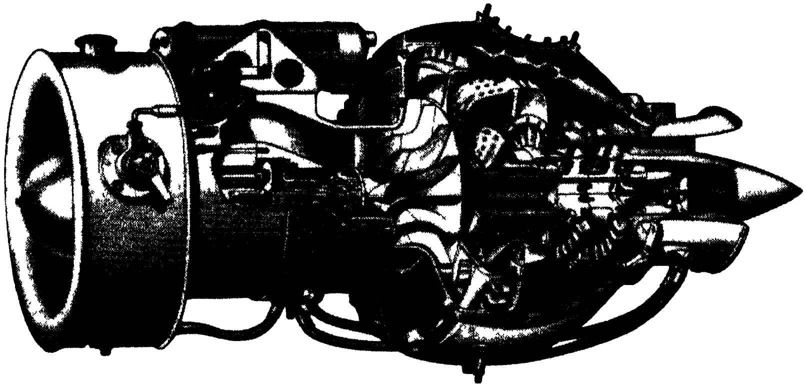


Fig. 12 . 15– Palouste turbine air compressor.

Units in another group function solely as air compressors and, apart from varied industrial applications, may be employed in aviation for pressurizing, air-conditioning, and supplying air to the reaction-propelled rotor blades of helicopters. Pimedon and Palouste units delivering air at 41·2 lb/in^2 pressure at 15° C at rates of 82 and 139 lb/min respectively are virtually simple jet units fitted with an oversize compressor. The air delivery is passed through a perforated combustion chamber casing to an enveloping container and ducted off.

A third unit, Arrius I, is of a different type and comprises an Artouste I single-shaft turbine driving an independent centrifugal compressor delivering air at the rate of 167 lb/min at 31·6 lb/in^2 pressure at 15° C.

The by-pass unit, Aspin II, falls in a special category, having performance characteristics intermediate between those of the turboprop and the turbojet. Assuming identical main components—compressor, combustion system and turbine—and the same air mass flow and consumption of fuel, a comparative estimation of the three types of power unit can be made.

It will be found that under static and low-speed conditions the by-pass

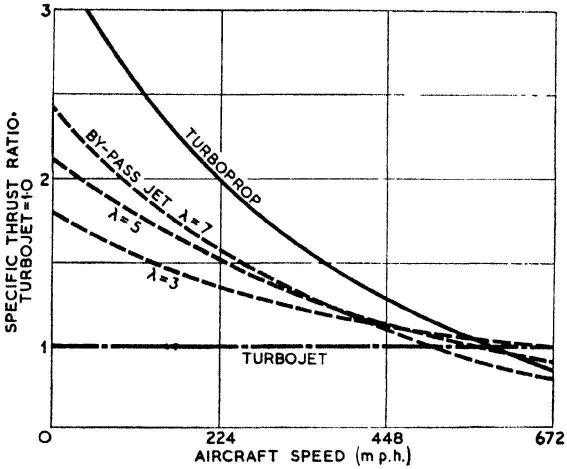


Fig. 12 : 16—Comparative curves of specific thrust. Three dilution ratios are shown for the by-pass unit.

unit develops greater thrust than the turbojet but less than the turboprop. As higher flight speeds are attained, however, the situation tends to reverse and the by-pass unit eventually develops more thrust than the turboprop but less than the turbojet. The relationship varies with aircraft speed and with the dilution ratio λ of the by-pass unit. This is the ratio of secondary to primary air mass flows and the curves in Figs. 12 : 16 and 12 : 17 are drawn for λ values of 3·0, 5·0, and 7·0.

The propulsive efficiency of the turboprop is superior to that of the by-pass unit at speeds up to 530 m.p.h., but only markedly so in the range 300 to 400 m.p.h. It will be noted that if the dilution can be varied

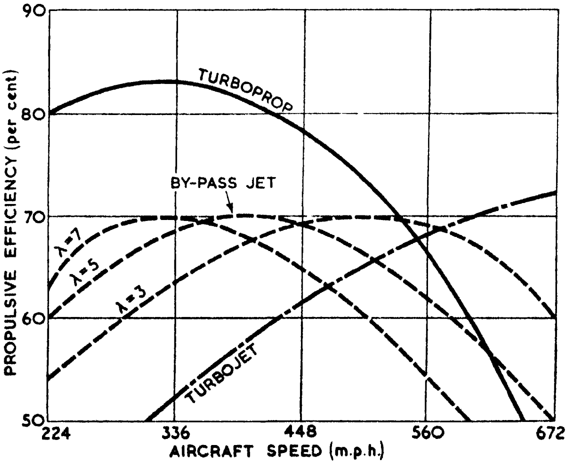


Fig. 12 : 17—Propulsive efficiencies of three types of turbine units against aircraft speed.

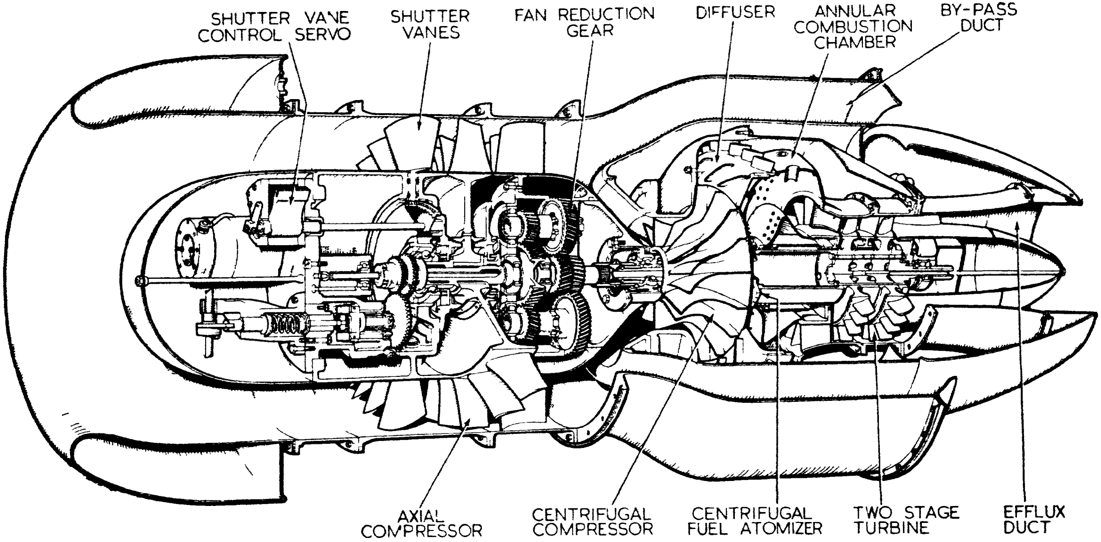


Fig 12 18—Turbomeca Aspin II by-pass unit the maximum propulsive efficiency of the by-pass unit can be arranged to occur at a desired aircraft speed which may be considerably higher than that at which the airscrew functions at its optimum efficiency. At aircraft speeds exceeding about 560 m.p.h. the turbojet becomes more efficient.

Between the nominal limits of operating speeds of 400 and 550 m.p.h. it would seem the by-pass unit has a field of application for civil and transport aircraft. It confers the incidental advantages that accrue from the elimination of the airscrew—reduced weight, simpler installation and lighter undercarriage—and gives better take-off and low-speed performance than the turbojet. The secondary air flow lends itself ideally to the provision of auxiliary fuel burners which can be used to furnish an increase of thrust for take-off or in emergency.

In Aspin II the ratio of dilution is controllable to establish optimum propulsive efficiency over a range of aircraft speeds. Variable-incidence shutter vanes, interconnected with the turbine speed control, regulate the proportion of the available energy developed by the turbine that is absorbed by the single-stage axial compressor. This method of control enables substantial variation to be effected instantly without the need to overcome the inertia of the rotating assembly. Despite the fact that the turbine is coupled to the axial compressor through the medium of a double reduction gear it can, if necessary, be suddenly unloaded. Furthermore, it facilitates the problem of starting. With the shutter vanes sufficiently closed, the torque resistance is substantially lowered and a conventional electric starter motor is adequate for the task. Thus the shutter vanes furnish a novel method of control which, with simultaneous regulation of fuel supply, gives an effective adjustment of performance to suit differing flying conditions. In effect, it is analogous to controlling the pitch of an airscrew.

Referring to Fig. 12 : 18, it will be seen that after passing the axial compressor the air is divided into two separate flows. The primary flow is to the centrifugal compressor and thence to the annular combustion chamber. Here it receives the energy released from the fuel and is then expanded through the two-stage turbine and discharged as a propulsive jet through a diffusing efflux duct. The design of this duct will determine the static pressure behind the turbine and consequently, to some degree, the division of energy between the axial compressor and the jet. It offers, therefore, the possibility of modifying the performance characteristics of the unit to meet specific applications.

Downstream of the axial compressor, the secondary flow is accelerated in a duct enshrouding the turbine unit and leading to the orifice of the efflux duct to effect a mixing with the primary flow.

To realize the optimum conditions of operation there is a co-ordinated control of the two variables, rate of fuel supply and position of the shutter vanes. Two differing régimes, economical cruising and flight under circumstances calling for rapid manœuvre, may be taken to indicate how this is effected. The control shaft carries actuation cams and is permitted both rotative and axial movement. In the first case, the control lever is turned through an arc to establish a speed of rotation, which is maintained constant by a centrifugal governor, and to adjust the shutter vanes to give the best specific rate of fuel consumption. For the second condition, depression of a control button brings into operation a hydro-electric servo-motor which moves the control shaft axially and thus ensures that speed of rotation is held to maximum by controlling the action of the centrifugal governor. Rotative movement of the control lever then influences solely the position of the shutter vanes, making possible a rapid variation of thrust. Starting is effected at the control condition for economical cruising.

As long as the speed of rotation is less than 70 per cent of the maximum, the shutter vanes are held stationary, the centrifugal governor is ineffectual and the fuel supply is regulated solely by movement of the control lever. Only at rotational speeds in excess of 70 per cent of maximum does the simultaneous control of speed and shutter vane movement come into operation.

The prototype, Aspin I, developing a maximum thrust of 440 lb, successfully completed a 1,000 hr test at speeds from 33,500 to 36,500 r.p.m. in January 1951. The current version, Aspin II, is rated at 795 lb thrust for take-off, and a continuous maximum s.l.s. of 662 lb with a fuel consumption of 0·510 lb/hr/lb. Dimensions are: diameter 23·8 in, length 63·5 in and weight 306 lb.

The range of Turboméca units is produced in Britain by Blackburn and General Aircraft Ltd., in the U.S.A. by Continental Motors Corp., and also in Jugoslavia.

ITALY

Initially, British turbojets were imported to re-equip Italian forces. Fiat has acquired a de Havilland licence and is building the Ghost. Both the Fiat and Alfa-Romeo companies, it is understood, will in the future produce axial-flow turbojets of either British or American design.

SWEDEN

Royal Swedish Air Force has made extensive use of British turbo-
in 1947 encouraged the design of experimental units by two
cerns. Svenska Flygmotor AB produced a centrifugal com-
and Svenska Turbinfabriks Aktiebolaget Ljungström
flow unit. Late in 1950 the two companies co-operated,
of the Air Force, to develop the STAL Dovern. The
alled for an output of 6,614 lb thrust at a specific

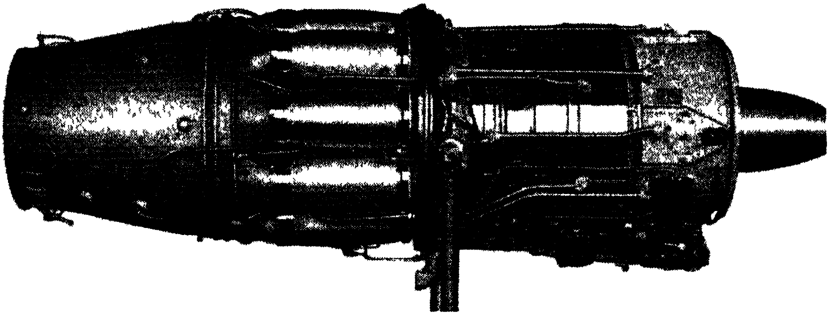


Fig. 12 : 19 STAL Dovern II turbojet, 7,250 lb thrust.

fuel consumption of 0 96 lb/hr/lb and a weight of 2,646 lb. This was well surpassed in the Dovern II which was developed, ready for series production, with a maximum static thrust of 7,275 lb, a consumption at normal maximum power of 0 92 lb/hr/lb and weight 2,632 lb.

However, for reasons of official policy, the Air Force terminated the programme and entered into agreement with Rolls-Royce to build the Avon under licence. The Flygmotor concern is already producing the de Havilland Ghost turbojet and Lucas fuel system components.

The Dovern II embodies modern practice and the specification includes a number of interesting features. Rotor, blades and casing of the nine-stage compressor are all of steel and four blow-off valves are fitted. It handles an air mass flow of 121 lb/sec and has a pressure ratio of 5 2 : 1. The nine tubular combustion chambers have flame tubes of Nimonic alloy and fuel is distributed on the Lucas full-range flow-control system. Jessop H46 steel is used for the turbine disc, to which the stub shaft is attached by registering lateral flanges secured by radially positioned conical bolts. For starting either a Rotax electrical starter or a B.T.H. cartridge-type turbo starter is mounted in the intake nose bullet. Each is used in conjunction with Rotax high-energy ignition units and Lodge plugs.

The II B version is furnished with a de-icing system in which the air intake and support struts, the inlet guide vanes and the nose bullet are heated by air tapped from the final stage of the compressor. Another, designated II C, is a II B unit equipped with an afterburner. STAL also have a project for a compound-compressor (two-spool) turbojet to develop 11,000 lb thrust.

SWITZERLAND

Even the most highly industrialized smaller countries would be faced with enormous economic difficulties in endeavouring to support the

EUROPEAN GAS TURBINES

Make and Type	*Comp. Stage and Type*	*Comb. Chambers*	*Turbine Stage*	*Max. Power Rating (Sea Level Static)*			*Dia. in*	*Length in*	*Weight lb*
				Thrust lb	*Shaft h.p.*	*Speed r.p.m.*			
				Germany					
BMW									
BMW 003A ..	7 A	Ann	1	1,760	—	9,500	27·0	125 0	1,344
Heinkel-Hirth									
HeS 011 ..	1 diag. 3 A	Ann	2	2,860	--	11,000	34·0	138 0	2,090
Junkers									
Jumo 004 B ..	8 A	6	1	1,890		8,700	30 5	152 0	1,585
				France					
Hispano									
Tay 253.. ..	1 C	9	1	7,700		12,300	49·9	100 0	1,972
S.N.E.C.M.A.									
Atar 101-D ..	7 A	Ann	1	6,614			36·1		1,969
Vulcain ..	7 A	Ann		13,000	-		45·6		3,080
Turboméca									
Piméné	1 C	Ann	1	220		35,400	16 1	34 6	122
Palas	1 C	Ann	1	350		34,600	15 9	47 2	159
Marboré ..	1 C	Ann	1	883		22,600	22 3	43 1	309
Orédon	1 C	Ann	2		158	35 400		34 6	179
Artouste I ..	1 C	Ann	2		276	33,400		36 6	183
Artouste II ..	1 C	Ann	2		394			31 5	200
Marcadau ..	1 C	Ann	2		400			86·5	307
Aspin II ..	1 A– 1 C	Ann	2	795	—		23 8	63 5	306
				Sweden					
STAL									
Dovern II ..	9 A	9	1	7,275	—	7,200	43·0	151 5	2,632

extensive research, testing, and production facilities necessary to ensure the continued development of aircraft turbine power units. Like Sweden, Switzerland made exhaustive studies and built experimental axial-flow turbojets, but eventually decided to produce a fully developed British unit. The Service Technique Militaire acquired a licence for the de Havilland Ghost and Sulzer Bros. Ltd. will manufacture and test.

CHAPTER 13

Compounded Units

FROM the earliest days of the internal combustion reciprocating engine the desire has been manifest to ' compound ' the cycle by a further expansion stage or stages, after the manner of the steam engine. A piston and cylinder mechanism, however, is relatively less satisfactory for expanding a large volume of gas at low pressure than a small volume at high pressure. As a consequence compounded piston engines did not attain commercial acceptance and in the conventional engine at least 50 per cent of the heat energy of the fuel consumed went to waste by way of the exhaust pipe and the cooling system. Less than 30 per cent was realized as useful power at the shaft.

The need for supercharging, to maintain the power output of aircraft engines at altitude, led to engines operating at higher pressures and substantially increased back pressures. Thus more energy was available in the exhaust and a logical step was to expand the gas in a turbine to drive the supercharging blower. An alternative was to use the exhaust energy, either by individual or collecter stacks, to furnish a supplementary propulsive jet—the so-called efflux propulsion system. Both methods were of practical value in improving the performance and economy of conventional piston engines.

It was apparent, however, that the energy in the exhaust was far more than was needed for auxiliary apparatus such as a supercharging blower and the surplus was then employed either mechanically to supplement shaft power output or as a propulsive jet from the turbine exhaust. Many projects on these lines have been suggested but very few plants have been actively developed and brought into practical operation.

This is the approach of using the exhaust gas turbine to augment the power output of an existing piston engine. Other than technical considerations may influence a decision to follow this course. It offered the possibility of at least a partial avoidance of supersession of the piston engine by the pure gas turbine. Fully-developed and proved engines and their components could be used ; existing plant, production equipment and techniques could be continued ; and, it was often suggested, a long period of development could be saved.

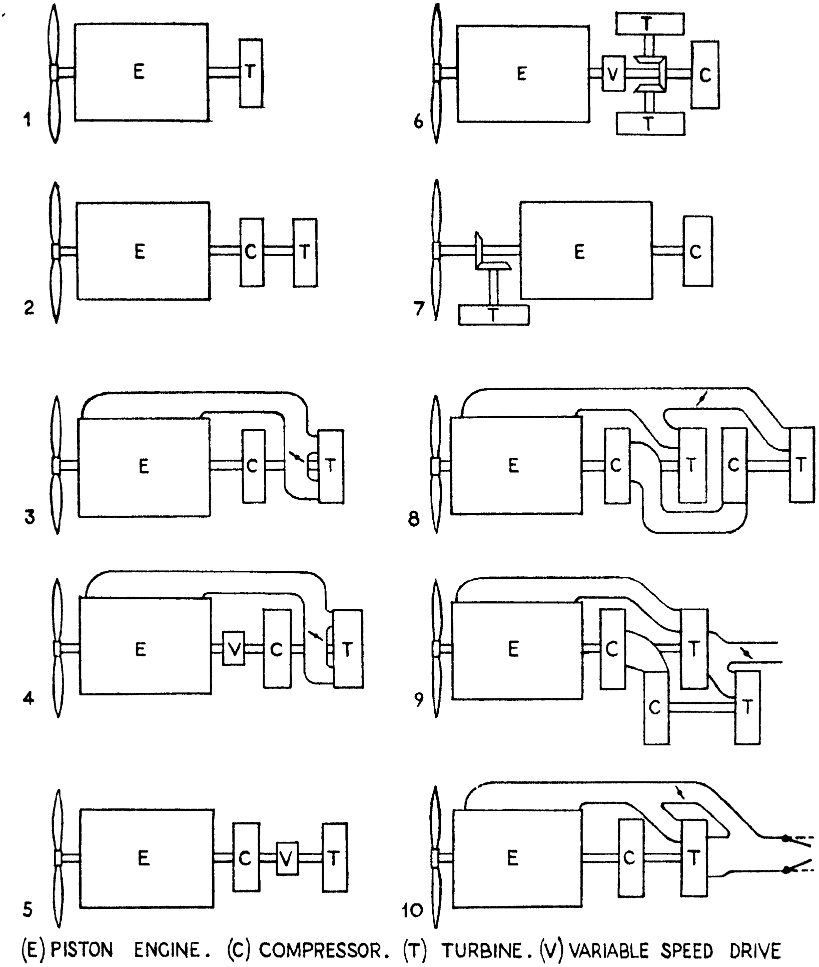
From a quite different standpoint is the other approach of improving the thermal efficiency, and thus the specific rate of fuel consumption, of the turbine unit by the combustion of fuel at a high instead of a low pressure. To this end a highly-supercharged piston engine is employed as a dynamic gas producer in place of the usual static combustion chambers. All shaft power developed by the engine is used to drive the supercharging compressor and all useful power output is delivered by the gas turbine. This necessitates the development of a piston engine designed specifically for such duty. If a rotary compressor is to be used probably a two-stroke diesel engine would be chosen but diesel-powered, free-flying piston compressors are also suggested.

A wide variety of component arrangements are possible, with resulting different characteristics for the complete plants. In a paper presented to the S.A.E. at Detroit, Mr. C. F. Bachle, Vice President in charge of research, Continental Aviation and Engng. Corp., listed the ten combinations shown diagrammatically in Fig. 13 : 1. Probably type 3 offers the most acceptable compromise of all factors for general application. For minimum fuel consumption over a wide range of operating conditions type 9 could be selected. In this assembly the two turbines are in series ; an arrangement giving maximum efficiency at high altitude. In type 10 the engine exhaust can be permitted at the highest aircraft speeds to by-pass the turbine and be discharged at approximately sonic velocity as a propulsive jet. At low cruising speeds the combined turbine and airscrew efficiency is about twice that for the jet. His estimate of the relative characteristics of the different arrangements follows :

Type	*Speed Range*	*Weight*	*Altitude Performance*	*Complication*
1	Poor	Good	Poor	Good
2	Poor	Good	Medium	Good
3	Good	Good	Good	Good
4	Good	Good	Good	Good
5	Good	Poor	Medium	Fair
6	Good	Fair	Good	Medium
7	Poor	Medium	Medium	Medium
8	Excellent	Poor	Good	Poor
9	Excellent	Poor	Excellent	Poor
10	Good	Good	Good	Fair

In America most effort has been expended in utilizing exhaust energy to increase the output and lower the specific fuel consumption of well-established piston engines in order to improve the performance of aircraft at all altitudes and to extend their operational range. Britain, on the other hand, has so far been primarily interested in the development of completely new compounded units to achieve the utmost fuel economy and thus extreme aircraft range. This, inevitably, is a much longer process.

In both countries there is some division of opinion as to whether the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(E) PISTON ENGINE. (C) COMPRESSOR. (T) TURBINE. (V) VARIABLE SPEED DRIVE

Fig. 13 : 1 - Compounded units with various component combinations.

compounded unit is economically justifiable and whether it is of transitory or permanent value. One school of thought holds that the piston engine and the turbine are inherently suitable for conjoint operation and thus merit development as a type, rather than as a modification of either. This view was succinctly expressed by Dr. H. R. Ricardo as follows : 'The piston engine is eminently suitable to deal with relatively small volumes at high pressure and temperature and the turbine, by virtue of its high mechanical efficiency and large flow areas, to deal with large volumes at low pressures. Clearly the logical development is to combine the two in series to form a compound unit.' The opposing view is that the potentialities of the compounded unit are limited and that the necessary developments of the turbine to exploit these potentialities fully will also be available to the turbine propulsion unit where they will prove of greater advantage.

Systems of exhaust gas turbines feeding power back to the engine shaft may, in general, be divided into two main classes, although there are a number of variations possible and a combination of the two classes has been tried. In the ' blow-down ' system individual pipes conduct the exhaust directly from the engine cylinders to the turbine entry. Thus the gas can expand relatively freely and there need be no increase in exhaust gas pressure. This avoids adverse effects on engine power output and cooling. In the alternative ' steady-pressure ' system the exhaust gases of a number of cylinders are delivered to a common chamber or duct in which a pressure is maintained, and thence pass to the turbine. Obviously, the pressure in the engine exhaust ports cannot fall below that in the chamber and, to some extent, a back pressure is exerted on the engine.

The ' blow-down ' system shows to advantage at low altitudes and the ' steady-pressure ' system is preferable for high altitudes. It would seem a ' steady-pressure ' system can best be exploited on an engine specially designed or adapted for the purpose whilst the ' blow-down ' turbine can more effectively be installed on an existing fully developed engine. The use of ' blow-down ' and ' steady-pressure ' turbines in combination can realize good performance at both low and high altitudes and at the same time maintain a low rate of fuel consumption. At 30,000 ft a consumption of 0·32 lb/b.h.p./hr has been quoted for an experimental engine. With this method the efflux from the ' blow-down ' turbine is collected and delivered to the ' steady-pressure ' turbine which may be geared back to the engine or used to drive a first-stage supercharging blower or other auxiliary unit.

Early in 1944 Allison commenced a development programme for a two-stage supercharged engine, designated V-1710-127, with a geared exhaust-driven turbine feeding power back to the crankshaft. The basic engine had a mechanically driven ' engine-stage,' or second-stage, supercharger and an ' auxiliary-stage ', or first-stage, supercharger driven from the crankshaft through a variable-ratio hydraulic coupling. Compounding was effected (see Fig. 13 : 2) by the addition of a modified General Electric turbine driving through the hollow shaft of the first-stage blower to a train of gears on the engine side of the hydraulic coupling. All the engine exhaust gases pass through the turbine at all times without reference to the first-stage compressor which is controlled in the manner previously employed on the basic engine.

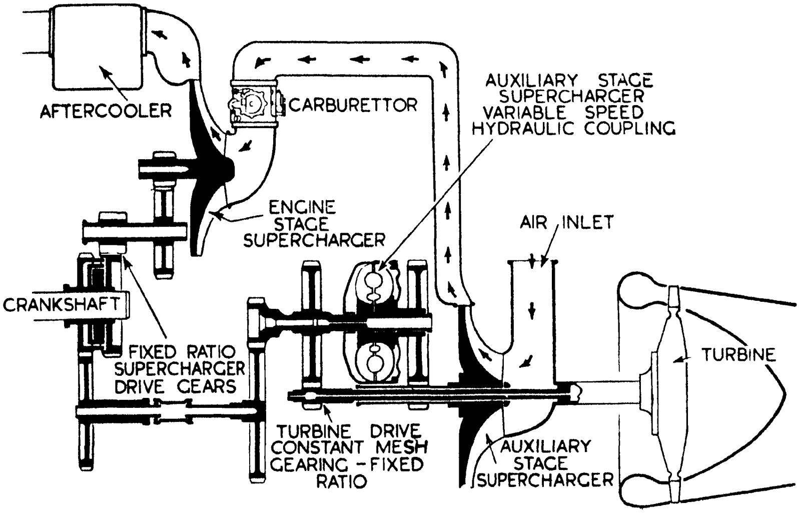

AFTERCOOLER
CARBURETTOR
AUXILIARY STAGE SUPERCHARGER VARIABLE SPEED HYDRAULIC COUPLING
ENGINE STAGE SUPERCHARGER
AIR INLET
CRANKSHAFT
FIXED RATIO SUPERCHARGER DRIVE GEARS
TURBINE
TURBINE DRIVE CONSTANT MESH GEARING - FIXED RATIO
AUXILIARY STAGE SUPERCHARGER

Fig. 13 : 2—Diagrammatic arrangement of turbine and drive on Allison V-1710.

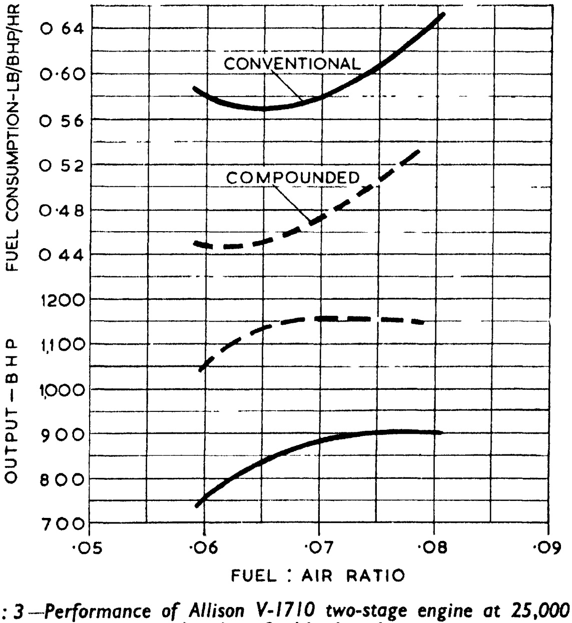


Fig. 13 : 3—Performance of Allison V-1710 two-stage engine at 25,000 ft with and without feed-back turbine.

Many tests under simulated altitude conditions were made and the curves in Fig. 13 : 3 show the results obtained at full throttle, 2,700 r.p.m., at 25,000 ft both with and without the feed-back turbine at a wide range of fuel : air ratios. The turbine was operated at a steady nozzle box pressure of 28 in Hg abs. At a fuel : air ratio of 0·066 the compounded unit showed a specific consumption of 0·45 lb/b.h.p./hr against 0·57 lb/b.h.p./hr for the conventional two-stage engine; a reduction of 21 per cent. Under the same conditions the power output was raised nearly 36 per cent from 825 to 1,120 b.h.p. The complete plant (Fig. 13 : 4) with two stages of supercharging, after-cooling, exhaust turbine, reduction gear and ducting, had a dry weight of 1,950 lb. Overall length was 97 in and the unit could be accommodated in the standard engine cowling diameter.

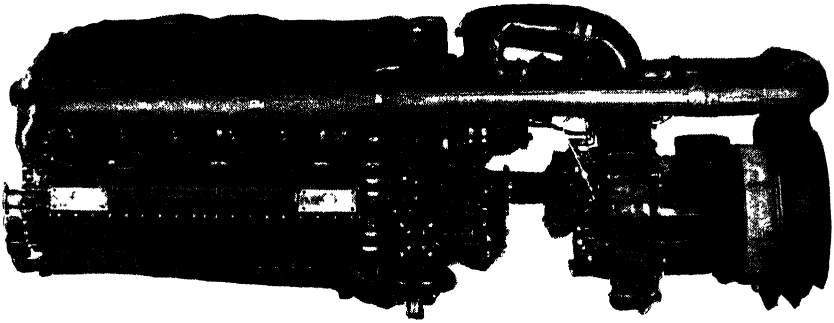


Fig. 13 : 4—Allison V-1710-127 compounded engine.

The turbine was designed for a maximum gas temperature of 940° C but early tests of the engine, operating with a manifold pressure of 100 in Hg, a fuel : air ratio of 0·08 and merely sufficient water-alcohol to suppress detonation, showed that the exhaust temperature would exceed this limit by a dangerous amount. In fact, a temperature of the order of 996° C was indicated under the most severe conditions of operation. To overcome this hazard an excess amount of water-alcohol was introduced but the quantity necessary was so great that power output was seriously reduced. Consequently, a water spray into the exhaust manifold to hold the gas temperature to 927° C was utilized. Operating with a fuel : air ratio of 0·067, a maximum water flow of 175 lb/hr was required to maintain a safe working temperature.

In 1946 the company, with the concurrence of the Air Materiel Command, took a decision to terminate the whole programme and concentrate its energies and resources on the development of turbojet and turboprop units. The reasons for this decision were disclosed by Mr. R. M. Hazen, Allison director of engineering, in 1947 :

'Had there been no revolution in types of power plants, this unit would have been revolutionary from the standpoint of specific weight, specific fuel consumption, cost per take-off h.p., or accepted criteria of aircraft engine evaluation. It could have been available for production in about two and a half years. We decided not to build this engine because concurrent studies of airscrew-turbine engines and experience with pure-jet units showed that going ahead with a two- or three-year development on such a compounded engine was a highly dubious venture. Basically, the weakness of such a development was that it was based on a turbine development in which the gas temperature entering the first-stage nozzle must of necessity be 982–1066° C with lengthy cruise operation in this temperature range.

'The engine was then compared with an airscrew-turbine unit on the basis of a reasonable exhaust temperature and also assuming the same maximum temperature as the compounded engine required. It was our conclusion that there was a great deal more certainty in developing a 871° C airscrew-turbine engine in a given time than in compounding a piston engine with a turbine requiring 982–1066° C temperature operation. On the basis of this comparison we recommended, and obtained, determination of our compounding development in order to devote our full engineering time to turbines of various types and, as time has gone on, we feel this decision was more and more sound and that the estimates made for airscrew-turbine performance were, if anything, conservative.'

In spite of this forthright declaration other American engine builders continued their developments with compounded units. Pratt and Whitney compounded the 28-cylinder Wasp Major radial engine with a General Electric CHM-2 two-stage turbine, driving a supercharging blower and discharging its efflux through a variable-area nozzle to furnish a propulsive thrust. No power was fed back from the turbine to the engine crankshaft.

To meet operating requirements, available energy in the engine exhaust was variably proportioned between the blower drive and the reaction jet and consequently control was relatively more complex than with feed-back turbines. Jet thrust to a maximum of about 800 lb was produced and the system was claimed to be suitable for operation at altitudes of 40,000 ft and over. A specific fuel consumption of 0·38 lb/b.h.p./hr was reported, giving about 20 per cent increased range for the same fuel load. This development programme, it is understood, has now been discontinued.

The Wright Cyclone 18-cylinder radial engine delivers about 2,700 b.h.p. at 2,900 r.p.m. for take-off. By compounding with three 'blow-down' turbines, take-off power output has been raised to 3,500 b.h.p. Despite the additional weight of approximately 540 lb, the increase in output lowers the specific weight from 1·06 lb/b.h.p. to about 0·97

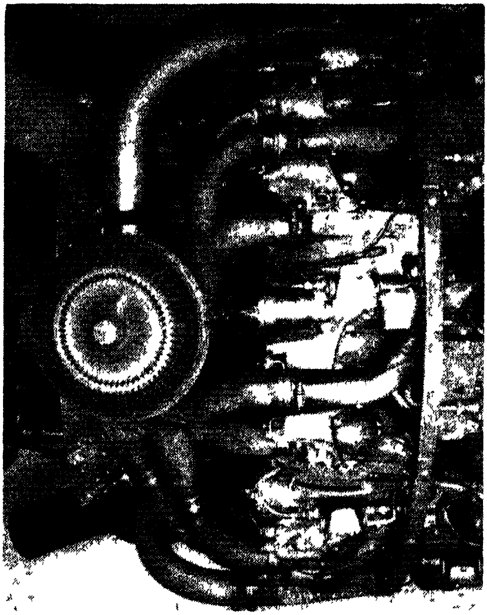


Fig. 13 : 5—One of the three turbines on the Turbo-Compound The pipe entering the casing on the extreme left conducts cooling air to the rotor assembly.

lb/b.h p. The specific fuel consumption is reduced from 0 72 lb/b.h.p /hr to 0 65 lb/b.h p./hr at take-off power, but at cruise rating at 20,000 ft the specific consumption falls to 0 39 lb/b h.p./hr, representing a 20 per cent improvement as compared with the standard engine. A special military version having a dry weight of 3,404 lb is rated to deliver 3,700 b.h.p , thus lowering the specific weight to 0 92 lb/b.h.p.

The three exhaust gas turbines are mounted on an extension of the crankcase to the rear of the cylinder banks and drive through radial shafts, bevel reduction gears, hydraulic couplings and final spur reduction gears to the crankshaft. To minimize ducting and radiation losses the turbines are located as closely as possible to the cylinders. Although pairs of individual exhaust pipes are joined before reaching the turbine, the phase relationship of the respective cylinders is such that no overlap occurs. Turbine rotors are approximately 11 in diameter with blades of Stellite welded to the disc. Exhaust gas reaches the turbine at supersonic velocity and with a temperature of 760–815° C. Rotor speed at maximum engine speed of 2,900 r.p.m. is 23,200 r.p.m. The complete plant is remarkably compact as it is only a few inches longer than the standard unit and is contained within the standard cowl lines.

The Wright Turbo-Compound has for some time been in service as the standard power installation for the Lockheed P2V-4, 5, and 6 Neptune anti-submarine patrol bomber and the Martin P5M-1 and 2 Marlin patrol flying boat. It is scheduled for installation in the new Super Constellations to serve as transports for both the U.S.N. and U.S.A.F. with a cruising range exceeding 5,000 miles at 325 m.p.h. It is also in airline service in the Douglas DC-7 and the latest Super Constellations.

The reduced specific rate of fuel consumption implies a 20 per cent increase in range for an aircraft carrying the same fuel load. Conversely, the craft could be operated over the same range with 20 per cent less fuel. In long range aircraft, in which fuel load can amount to 2 5 times the payload, such a saving of fuel would make possible a 50 per cent increase in payload. Further advantages listed are :

1. Efficient operation from sea level to high altitude.
2. Blow-down turbines impose a minimum back pressure on the engine cylinders and consequently engine output is not impaired and cooling is not seriously affected.
3. No additional controls are necessary.
4. The turbines act as silencers.

An example of a compounded plant designed as a whole and not modified from existing units is shown diagrammatically in Fig. 13 : 7. This is the subject of a patent by the British company, D. Napier and Son, Ltd. The reciprocating engine A, operating on the two-stroke cycle with compression ignition, drives through gearing at the forward end an airscrew B arranged with its axis parallel to the engine crank-

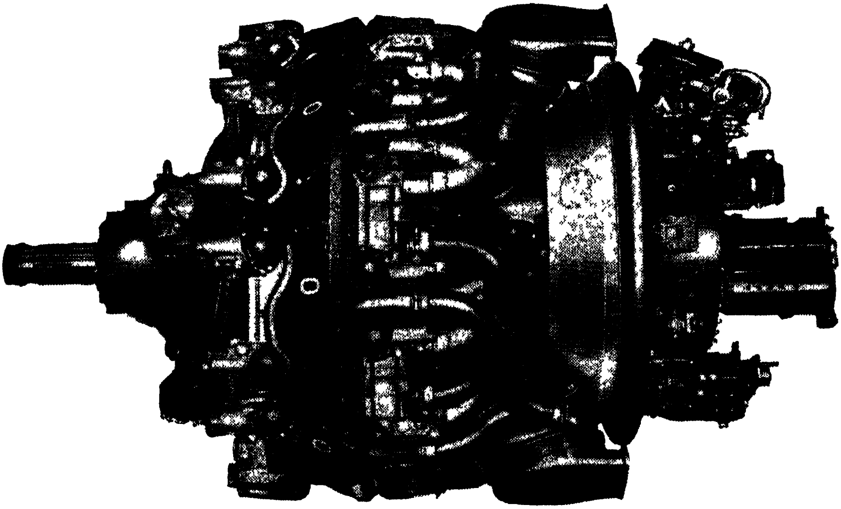


Fig. 13 : 6—Wright Turbo-Compound with three blow-down turbines.

shaft. A centrifugal compressor C at the rear is driven from the forward end of the crankshaft by means of a torsionally flexible shaft and gearing. The gases produced by the engine are delivered to a duplex turbine unit having main and secondary rotors, D and E respectively, mounted on a common shaft. An extension of the turbine shaft passes through the hollow impeller shaft of the centrifugal compressor to drive the axial-flow compressor F and in turn, through suitable gearing, the co-axial airscrew G. Air delivered by the axial compressor is passed through an intercooler H before reaching the centrifugal compressor and arrangements are made whereby the cooler may be by-passed or alternatively the supply of coolant may be varied.

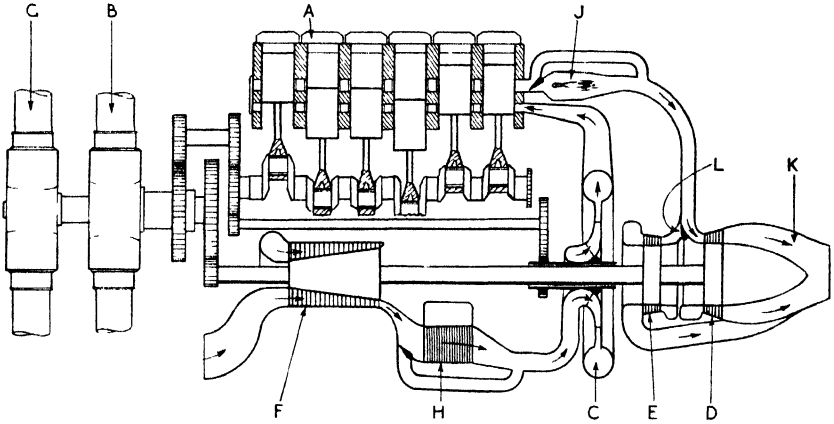


Fig. 13 : 7—Napier project for compounded unit with two-stroke compression-ignition engine and reheat chamber.

Between the engine and the turbine there is arranged a combustion chamber J into which fuel may be injected and burned. A duct may be provided to by-pass the chamber or the exhaust gases may traverse the chamber at all times and control be effected by regulation of the supply of fuel.

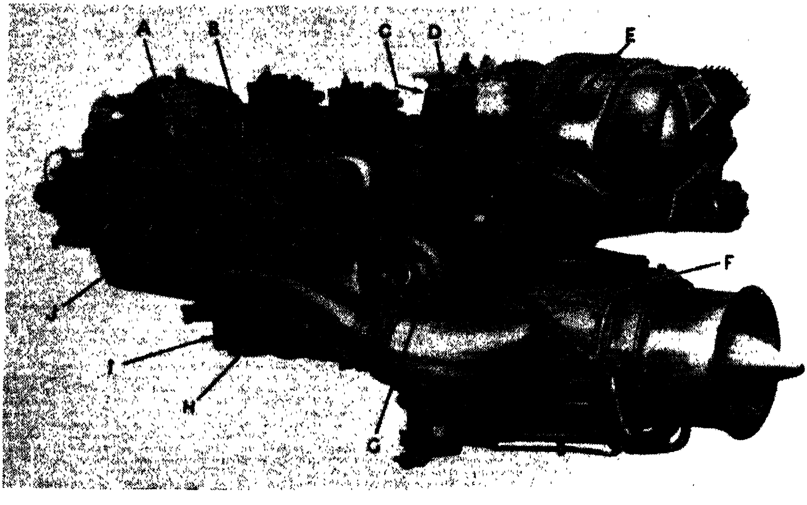
The main flow of the gas is directed through turbine D and then issues as a jet with reactive effect from nozzle K. A valve L enables some portion of the gases to be diverted as and when desired and passed through the secondary turbine E to supplement the power developed by the main turbine in driving the axial flow compressor and the airscrew.

The first prototype Napier compounded unit, the E.125, was exhibited at the S.B.A.C. Farnborough Display in 1951. It comprised a horizontally-opposed, 12-cylinder, liquid-cooled, compression ignition, valveless two-stroke engine driving one half of a counter-rotating airscrew. The engine was supercharged by an axial compressor and a centrifugal compressor in series, and the exhaust gases were led to a gas turbine driving the other half of the airscrew. To provide maximum power for take-off, a reheat chamber in which additional fuel could be burned was incorporated in the exhaust ducting from each bank of cylinders. When reheating was employed a secondary turbine was brought into operation.

Take-off power rating was 3,125 e.h.p., obtained as 3,000 s.h.p. and 320 lb thrust. At that rating the specific fuel consumption was 0·36 lb/e.h.p./hr. Either aviation kerosine or diesel fuel could be used and a changeover could be effected without any adjustment to the engine. With overall dimensions of 126·5 in long, 58·25 in wide and 49·25 in deep, the dry weight was 4,200 lb.

The E.125 was extensively test-flown mounted in the nose of a Lincoln bomber. It served as a test and experiment unit and accumulated the operating experience upon which was based the design of a later power plant, shown in Figs. 13 : 8 and 13 : 9.

The later E.145 Nomad represents a considerable advance and is simpler and more economical to produce. As compared with the original unit it has a single axial air compressor instead of an axial and a centri-



A. Front engine mounting.
B. Fuel injection pump.
C. Electric starter mounting.
D. Rear engine mounting.
E. Turbine/crankshaft gearbox.
F. Multi-stage turbine.
G. Coolant pump.
H. Axial-flow compressor.
I. Air intake.
J. Ignition for starting.

Fig. 13 : 8—Napier E.145 Nomad compounded unit with variable-ratio drive between turbine and engine crankshaft.

fugal unit, no reheat chambers, no secondary turbine, and a single instead of co-axial airscrews. Additionally, a stepless variable gear of the Beier type is incorporated to adjust automatically the ratio of the reduction gearing coupling the turbine to the engine crankshaft.

Although the static rating has been raised to 3,135 e.h.p., the frontal area is reduced from 19·92 to 19·25 ft^2 and the overall length, including the jet tail cone, is now 119 in. Dry weight is down to 3,580 lb, giving a specific weight of 1·142 lb/e.h.p., but the installed weight is lowered by half a ton. Over a wide range of altitudes and speeds the specific fuel consumption remains between 0·33 and 0·35 lb/e.h.p./hr.

The 12-cylinder diesel engine has a bore and stroke of 6·0 in and 7·375 in respectively, giving a swept volume of 41·1 litres. Volumetrically the cylinders have a compression ratio of 8 : 1. They are fed with air by the 12-stage axial compressor, mounted below the crankcase, which has an air mass flow of 13 lb/sec with a maximum pressure ratio of 8·25 : 1. Under take-off conditions the boost pressure is 89 lb/in^2 abs and the effective overall compression ratio is 31·5 : 1. The engine operates with a b.m.e.p. of 205 lb/in^2 and the overall expansion ratio is 24 : 1. As higher altitudes are reached the overall expansion ratio is increased and the turbine-compressor unit is speeded up, relative to engine speed, to increase the boost pressure and maintain power output. Concurrently the overall efficiency is improved and this is reflected in the lower specific fuel consumption at altitude.

Take-off power of 3,135 e.h.p. at sea level is obtained with engine and compressor speeds of 2,050 r.p.m. and 18,200 r.p.m. respectively and a fuel consumption of 0·345 lb/e.h.p./hr. When cruising at 300 kt at sea level about 1,860 e.h.p. is developed at speeds of 1,750 r.p.m. and 14,960 r.p.m., giving a boost of 43 lb/in^2 abs, with a fuel consumption of 0·355 lb/e.h.p./hr. At the recommended cruising conditions—300 kt T.A.S. at 25,000 ft—the speeds are 1,750 r.p.m. and 19,600 r.p.m. producing 2,024 e.h.p. at a specific consumption of 0·327 lb/e.h.p./hr.

Take-off power may be increased to 3,580 e.h.p. by water injection into the engine inlet manifolds, corresponding to an output of 87·1 e.h.p./litre and a weight of 1·0 lb/e.h.p. Further augmentation of power output may be obtained by reheating. A small quantity of fuel is burned in the exhaust manifolds between the engine and the three-stage turbine, there being an adequate amount of excess air in the exhaust gases to support combustion. Using both water injection and reheat, take-off power can be raised to 4,095 e.h.p., which is specifically equivalent to 99·6 e.h.p./litre and a weight of 0·87 lb/e.h.p.

Studies made by the Napier Co. of the comparative performance of hypothetical aircraft powered by different types of power unit are of particular interest. All carrying the same payload of 25,000 lb over a

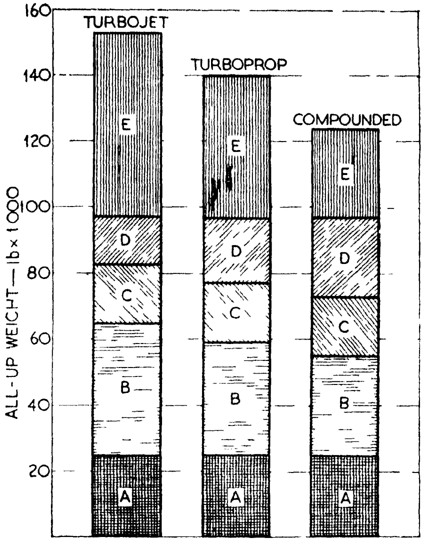


Fig. 13 : 10—All-up weights of aircraft with different power plants. Payload 25,000 lb. Still-air sector distance 2,860 miles.

A Payload.
B Structure
C Crew and equipment.
D Power plant
E Fuel

still-air sector distance of 2,860 miles, the first has four turbojets of 10,000 lb s.l.s. thrust, the second four turboprops of 3,780 e.h.p. for take-off and the third Nomad compounded units of 3,135 e.h.p. for take-off Due to the low specific rate of consumption the Nomad powered aircraft needs to carry less fuel and reserves and consequently the same payload can be carried in a smaller airframe, as illustrated graphically in Fig. 13 : 10. Since the cost of an aircraft is approximately proportional to equipped airframe weight the compounded unit claims to offer a substantial reduction in capital cost in addition to operational economy. The estimated costs of the three aircraft are £623,000, £520,000 and £491,000 for turbojet, turboprop and compounded unit types respectively.

Fuel cost is the largest individual item in the direct operating costs of an aircraft. On the common basis of aviation kerosine, the fuel costs of the three aircraft studied will be 35 per cent, 28 per cent and 22 per cent respectively of direct operating costs. If the aircraft is powered by compounded units running on diesel fuel the figure is lowered to 17 per cent.

A low specific rate of fuel consumption confers potentially all the advantages of long range. Thus it is conducive to regularity of operation, wider choice of airports, and economy by avoiding the need for station maintenance and operational personnel, spares, and refuelling facilities at intermediate landing points. The cumulative effect of these factors, if they permit a particular sector to be flown non-stop, can largely offset the speed advantage of a competitive aircraft. This is indicated in Fig. 13 : 11 in a graph showing the elapsed times for a turbojet aircraft cruising at 530 m.p.h. at 40,000–45,000 ft and a Nomad-powered aircraft cruising at 353 m.p.h. at 25,000–30,000 ft on the

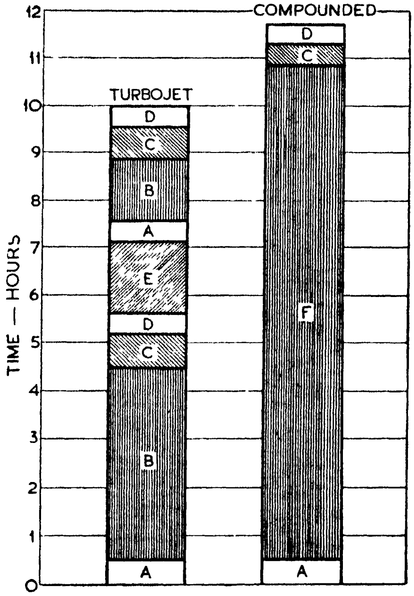


Fig. 13 : 11—Elapsed times for aircraft with turbojet and compounded power plants on London—New York sector.

A. Climb.
B. Cruise at 530 m.p.h.
C. Descent.
D. Circuit, approach, land.
E. Customs, immigration, refuel.
F. Cruise at 353 m.p.h.

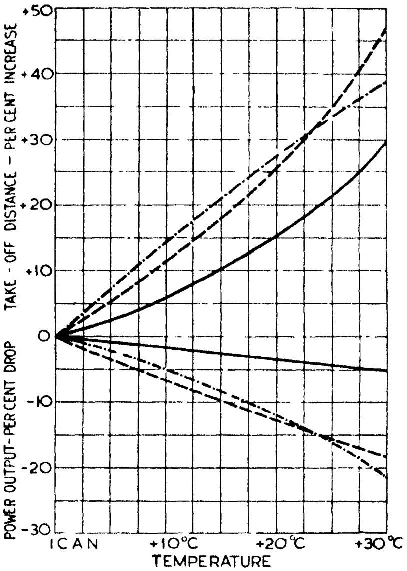
London–New York sector. The turbojet, perforce, refuels at Gander and takes 10 hr while the Nomad-powered aircraft flies non-stop in 11·7 hr on a much lower consumption of fuel.

All types of engines consuming air suffer from a decline in power output with increase of ambient air temperature. In tropical or sub-tropical countries this leads to the necessity for long runways for take-off or special boosting equipment for the engines. Wet compression, afterburning, and liquid rockets are under consideration for jet aircraft. In this respect the compounded unit shows to advantage over both turbojets and turboprops. As shown in Fig. 13 : 12, a temperature rise of 15° C above I.C.A.N. conditions results in a reduction of about 2·5 per cent of the power output of the compounded unit as compared with approximately 9·0 per cent for turbojet or turboprop.

The effect on take-off distance, expressed as a percentage increase on the distance required under I.C.A.N. conditions, is also shown. Assuming in all cases that one of the four engines fails at the critical point, the aircraft with compounded units will only require 10 per cent extra distance while turbojets or turboprops need about 20 per cent for a temperature rise of 15° C.

Of the many and varied schemes of compounding that have been proposed a large number had the primary aim of increasing the power output of an existing piston engine. In the quest for higher power output, however, the straightforward turbojet or turboprop unit offered better prospects. The justification of the compounded unit as a type will probably be found in the advantages accruing from its enhanced overall thermal efficiency. Apart from increased power output for take-off and

Fig. 13 : 12—Power fall-off and increase in take-off distance with rise in ambient air temperature.



economical operation while cruising, the necessary fuel load can be adjusted either for increased range or increased payload.

For long-range civil operation specific fuel consumption is liable to be the decisive factor in determining if a service is economically feasible. In some instances, if a reasonable payload is to be carried, it becomes necessary to set up refuelling bases and the operator is then faced with the cost of transporting fuel to the base. Substantially increased aircraft range offers the possibility of re-routing, fewer stops and a reduction, at least, of special refuelling bases.

For extreme range, as for instance in military service, speed becomes a factor and power units of other type may be necessary to obtain the lowest gallons-per-mile figure. On the other hand, the low specific consumption of the compounded unit confers unsurpassed endurance and thus makes it the most suitable type for naval patrol and search duties in which the lowest gallons-per-hour figure is the criterion.

Commercially, the compounded unit would appear to be admirably suited for the economical transport of passengers or goods at medium speed and medium altitude, say, 350 m.p.h. at 20,000–25,000 ft. Running on a relatively ‘ safe ’ diesel fuel, the engine can be of robust construction and the turbine unit readily demountable for servicing. It could provide the basis for a genuine ‘ tourist ’ type of passenger service.

CHAPTER 14

Ramjets, Pulsejets and Rockets

JET propulsion ducts, of both continuous and intermittent-impulse types, have for long engaged the attention of scientists and engineers. Research and experimental operation of thermal ducts, in fact, preceded consideration of the turbojet. Relative to turbine units they offer the possibility of cheap and rapid construction, light weight, simplicity of operation and high speed—although at a higher rate of fuel consumption. These considerations pointed to their potential value for short-range, piloted interceptor aircraft and, in particular, for expendable pilotless missiles. Actually, development for the latter purpose was accelerated by the Germans in the Second World War and resulted in the large-scale production of a pulsejet unit for the notorious V-1 flying bomb with which they bombarded London from the European mainland.

RAMJETS

A continuous thermal duct is mechanically the simplest, yet aerodynamically the most subtly complex, form of jet propulsion unit. Formerly termed an ' athodyd,' a convenient contraction of aero-thermo-dynamic-duct, it is now known almost universally as a ramjet. Following earlier researches by Marconnet and Chanute, the principle of thermal duct propulsion was propounded by the Frenchman Lorin as long ago as 1913. At that time it was only of academic interest as the speed of current aircraft was far too low for the realization of ramjet propulsion and consequently little or no practical investigation was made.

Dependent upon air rammed into a diffuser duct, the ramjet requires a minimum velocity of about 300 m.p.h. for effective operation. An aircraft powered by a ramjet must necessarily be ground-launched with the aid of a catapult or rocket, or air-launched from a carrier aircraft travelling at the requisite speed.

As in all jet propulsion units, thrust is obtained by reaction to the change in momentum of the air flowing through the system. To this end the ramjet comprises three principal sections. At the forward end is a divergent duct, in the centre a combustion chamber where the fuel is burned to release heat, and at the rear a convergent nozzle. Air at atmospheric pressure is rammed at a velocity approximately equal to the airspeed of the aircraft into the diffuser intake duct which raises the pressure and lowers the velocity. Into the compressed air atomized fuel is injected and, after initial ignition by a spark plug, burned continuously to effect the first stage of expansion and consequent increase of velocity. Finally the air and combustion products are expanded down to approximately atmospheric pressure in the convergent nozzle section and are discharged at about twice the speed of the aircraft.

The matching of the three sections entails meticulous aerodynamic balancing. Relationship between the sections is the subject of continued experiment but with a fixed geometrical construction peak performance can be attained only at the design operating conditions. Satisfactory operation depends upon forward velocity and an increase in velocity raises efficiency as the improved ram pressure gives a higher combustion chamber pressure and consequently a better expansion ratio to ambient air pressure.

At a given altitude thrust horse-power varies as the cube of flight speed but an increase in altitude results in a decrease of thrust as the air mass flow is reduced. Thus the ramjet is determined as a means of propulsion for craft or missiles of high velocity at relatively modest altitudes. A pressure ratio of about 7·5 is obtained at an airspeed of 1,400 m.p.h., giving an efficiency approximately equivalent to that of a piston engine-airscrew power unit.

To achieve efficient operation over a wider range of speeds and altitudes, intake and exhaust nozzles of automatically variable area would be necessary. Combustion would present difficult problems as, within the confines of a short chamber, wide ranges of air : fuel ratios and air velocities must be possible. Metallurgy is a factor of some importance as exhaust temperatures of 1,650° C or more may be attained. Current experimental units are constructed of stainless steel and Inconel. Some have liquid-cooled jackets and some ceramic-lined combustion chambers.

Remarkable foresight was shown by the French engineer René Lorin who in 1908, as mentioned in Chapter 2, propounded a scheme for jet propulsion by intermittent impulses generated by a piston engine. By 1913 he had evolved designs for a continuous-flow thermal duct which, basically, is the ramjet of the present day. Up to recent times the ramjet was termed generically the ‘Lorin duct’ on the Continent.

In 1915 Lorin, who was an artillery officer, offered the French military authorities a project for the long-range bombardment of Berlin by ‘aerial torpedoes,’ These were winged missiles propelled by his thermal ducts—literally flying bombs—and carrying 440 lb of explosives. Control was to be effected by radio from aircraft stationed along the flight path. Rejected by the authorities as unrealistic, the scheme was not disclosed by publication until 1918.

The Leduc 0·10 ramjet aircraft was developed to the operational stage after the termination of the Second World War. A mid-wing monoplane of 34 ft 6 in span, the tubular fuselage forms the thermal duct with the

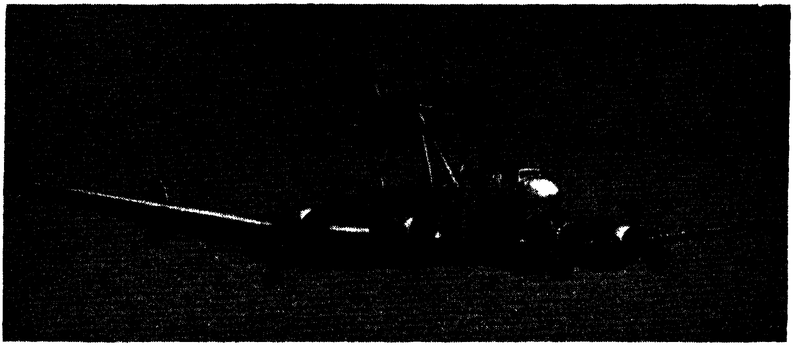


Fig. 14 : 1—The Leduc 0·10 ramjet aircraft was air-launched from a Languedoc 161.

dual-seated cabin suspended as a ' bullet ' in the entry section. The weight, with 250 gal of fuel, was 6,160 lb and the duct was calculated to develop a thrust of 3,500 lb at 560 m.p.h. at sea level. Carried aloft on the back of a Languedoc 161 it was first air-lauched, as a glider, in October 1947. In April 1949 it was launched at 10,000 ft and made a 12 min flight on half power, reaching a speed of 450 m.p.h. Speeds of about 500 m.p.h. and an altitude of 36,000 ft were attained in later flights also at half power.

Three of these experimental craft were completed but the fourth, now undergoing final tests, is about one-third larger. Designated Leduc 0·21, it is equipped with a Turboméca Marboré II turbojet at each wing tip to effect a take-off from the ground. The first supersonic Leduc is the model 0·22, a swept-wing prototype built largely of steel and intended for a Mach number of over 1·5. Flight trials are to begin in late 1955.

The first of the American ramjets to be ' declassified ' was the subsonic Marquardt C30-1·0, in which designation ' 30 ' signifies the diameter in inches and ' 1·0 ' the designed limiting Mach Number. Constructed by Marquardt Aircraft Company of Venice, California (a subsidiary of the Aerojet Engineering Corp.) to designs by the University of Southern California, this unit is 11 ft long and at Mach 1·0 at sea level is said to develop more than 20 t.h.p. per pound weight. Specific fuel consumption is fully four times that of a typical turbojet but, nevertheless, the C30-1·0 may prove efficient as a power unit for short-duration missiles in the design of which the light weight may offset the heavy fuel consumption.

From front to rear the essential components are (1) an inlet-diffuser section, in which the rammed air is decelerated to an appropriate velocity for combustion, with a corresponding conversion of kinetic energy to pressure energy; (2) a fuel injector which produces a finely atomized spray ; (3) a ‘ flame holder,’ in the form of a grid, which maintains stable combustion ; (4) a ceramic-lined combustion chamber and (5) an exit nozzle designed to accelerate the gases so that the products of combustion are exhausted at approximately ambient pressure and at a velocity about twice the flight speed.

An experimental installation of the C30-1·0 has been made in a Lockheed F-80 Shooting Star turbojet fighter, two units being attached at the wing tips. This machine was the first piloted aircraft to fly solely under ramjet power. Development was undertaken in the light of experience with a Mustang, likewise having tip-mounted Marquardt ramjets, though these were of an earlier 20 in diameter design.

Marquardt subsonic ramjets measuring 7 ft long and having a maximum diameter of 20 in have been extensively air-tested in PTV-N-2 (Gorgon IV) pilotless missiles built by the Glenn L. Martin Co. for the U.S. Navy. Although fuel for only 4½ min is carried, several remotely-controlled flights of more than 10 min duration have been recorded after launches from beneath the wing of a Northrop Black Widow. The Gorgon IV, a high-wing monoplane 22 ft long and 10 ft span, weighs 1,600 lb of which 700 lb is 80-octane petrol as fuel for the ramjet. Data relative to the performance of the ramjet is telemetered to a ground receiving station, together with details of speed, height, brake drag and controllability. A parachute can be set to operate automatically on cessation of powered flight and several Gorgons have been recovered so little damaged that only a small amount of work was necessary to prepare them for further flight. On the carrier aircraft, the weight of the Gorgon, suspended beneath the starboard wing, is counterbalanced by a tank under the opposite wing, filled with fuel. This fuel is pumped into the wing tanks of the Black Widow as fuel for the Gorgon's ramjet is transferred to the missile prior to launching.

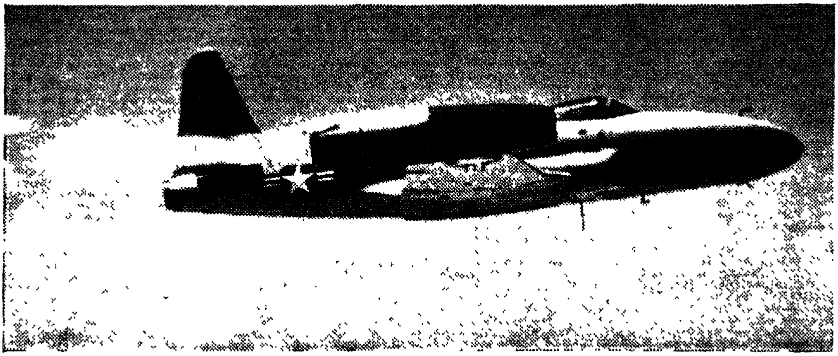


Fig. 14 : 2—Lockheed F-80 Shooting Star flying on two tip-mounted Marquardt subsonic ramjets.

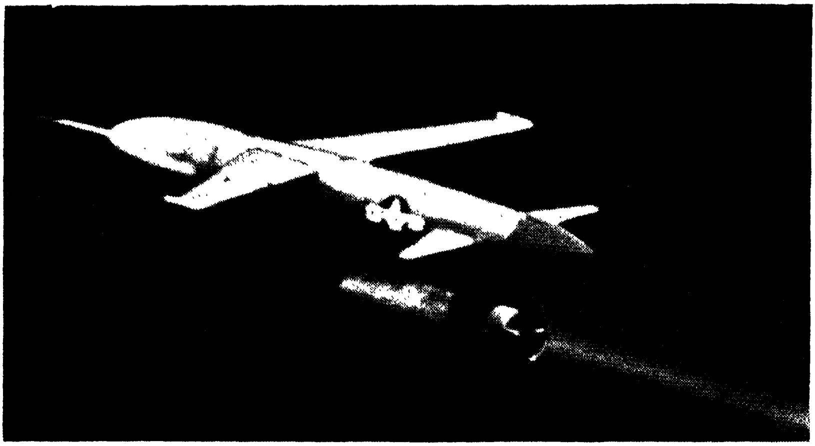


Fig. 14 : 3—Martin-built Gorgon IV ramjet missile for U.S.N.

An experimental ramjet 14 ft long and 16 in diameter has been utilized by the N.A.C.A. for supersonic tests up to Mach Numbers of 2·4. Originally drop-tested from a B-29, units of this type were flight-tested on a North American Twin Mustang. They were attached beneath the wings and were provided with a compartment, located in the inlet section, containing a radio transmitter, controls and fuel. In the nose is a long antenna for the transmitter and at the rear four fins to provide aerodynamic stability.

A supersonic ramjet, developed for the U.S. Navy by the Aerojet

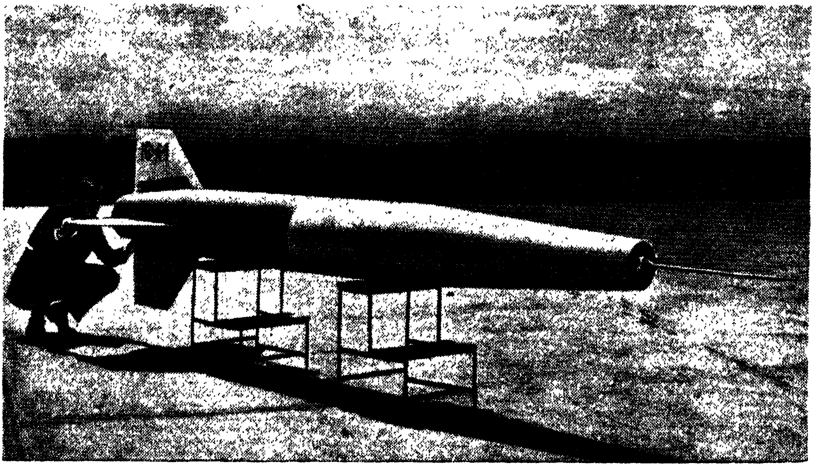


Fig. 14 : 4—N.A.C.A. 16 in supersonic ramjet.

Engineering Corporation, in conjunction with the Johns Hopkins University, has a diameter of 15 in and weighs some 150 lb. The designed thrust is 2,650 lb at 1,500 m.p.h.

In the supersonic wind tunnel at the N.A.C.A.'s Lewis Memorial Laboratory, Cleveland, ramjets have been tested in air velocities up to Mach 2 under conditions obtaining at 35,000 ft altitude. Additional facilities are available in the Aircraft Engine Research Laboratory of the N.A.C.A., where a wind tunnel permits operation of ramjets under simulated high-altitude supersonic conditions. A special test laboratory for ramjets has been established by the Wright Aeronautical Corporation at Wood Ridge, New Jersey. Some indication of the scale of units to be investigated may be gathered from the equipment which can deliver 6,000 gal/hr fuel and 140 ton/hr air. The steel test cell is 12 ft diameter and 96 ft long and flight conditions up to Mach 4 velocity and 80,000 ft altitude can be simulated.

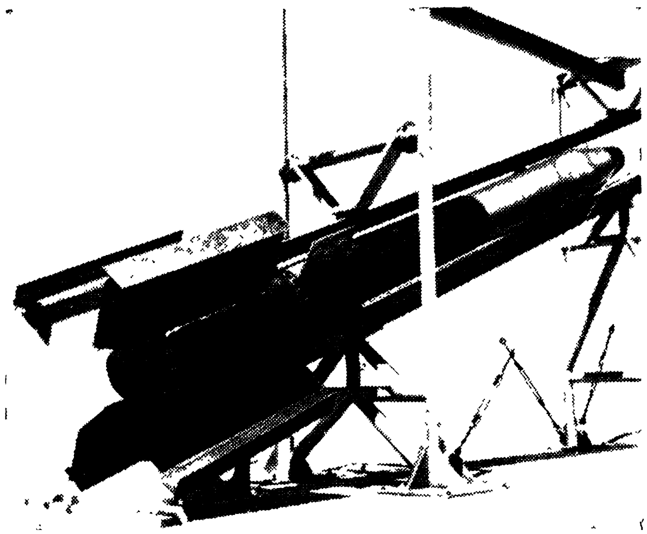


Fig. 14 : 5—U.S.N. supersonic ramjet with take-off rocket on the launching ramp.

The Germans were quick to appreciate the possibilities of the ramjet as a power plant for fighters, short-range bombers and guided missiles. Development was at first the concern of Walter at Kiel and of Dr. Saenger of the D.F.S., but whereas Walter did not progress beyond the model stage, full-scale ramjets of Dr. Saenger's design were flight-tested on aircraft. A unit of 2,400 e.h.p. was tried on Do 17Z and one of 20,000 e.h.p. on a Do 217E.

The results were not completely satisfactory. Combustion efficiency

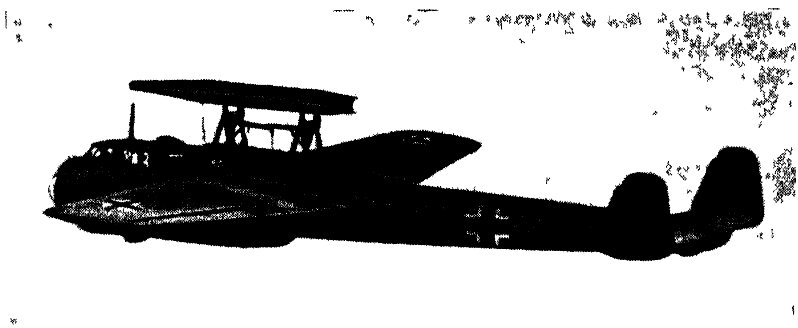


Fig. 14 6— Dornier 17Z fitted experimentally with a Saenger ramjet.

was too low, as evidenced by the length of the flame issuing from the duct, and designed thrust values were not attained. Nevertheless, during the war studies were made of a number of designs for ramjet interceptors and ' Blitz ' bombers

One of these projects, a fighter-bomber having a tubular fuselage forming the Saenger ramjet, is shown in outline in Fig 14 : 7. The duct was approximately 35 ft long and 8 ft diameter at its major section, with the compartment for a crew of two mounted externally above. The small fan in the entry diffuser, it is understood, was to induce air and promote a turbulence to ensure a static light-up, although rockets were provided

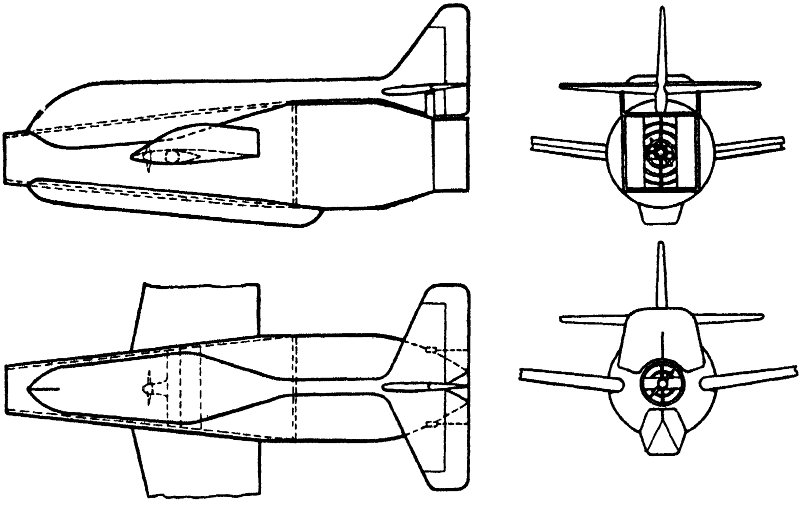


Fig. 14 . 7—Projected German ramjet aircraft.

for the take-off. It will be noted that a variable-area discharge nozzle is fitted, controlled by two vertical shutters.

With 5,280 lb fuel, a military load of 2,200 lb, take-off rockets weighing 1,543 lb, and crew, the total take-off weight was 15,432 lb. Performance was estimated to include a take-off and climb to 39,000 ft in 2·5 min, maximum speeds of 528 m.p.h. at sea level and 466 m.p.h. in the stratosphere, and a total flight range of 500 miles.

Focke-Wulf proposed a swept-wing fighter with two units of the Pabst type, more compact than the Saenger and promising remarkable efficiencies even at subsonic speeds. These were mounted at the tailplane extremities, and a Walter bi-propellant rocket would provide 6,600 lb take-off thrust.

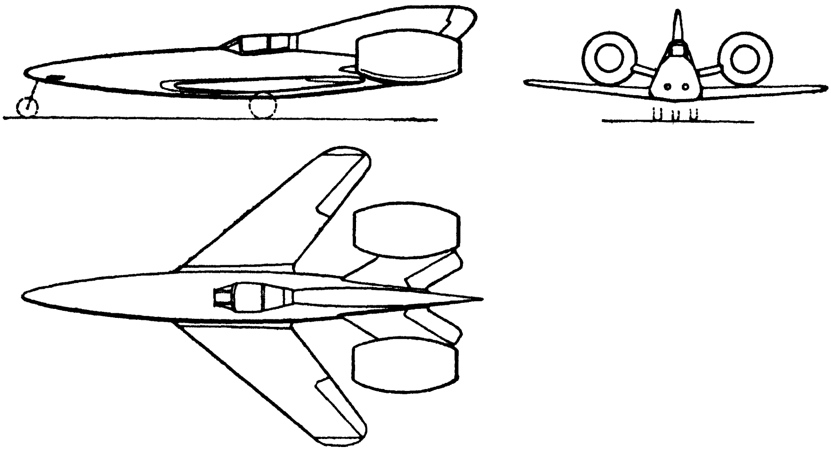


Fig. 14 : 8—Focke-Wulf fighter with two Pabst-type ramjets.

Every effort was made to reduce internal and external ' casing drag ' of the ramjets by careful aerodynamic design. Using the duct diameter as a reference value, the streamlined intake diffuser had a length of 1·0 diameter, the combustion chamber 0·75 diameter and the discharge nozzle 0·5 diameter, giving an overall length of 2·25 diameters. This contrasts sharply with the Saenger design which necessitates a length of 5 diameters. The area exposed to air friction is thus much reduced and the casing drag is decreased by about 50 per cent. The external dimensions of the unit were length 8 ft 10 in and diameter 4 ft 5 in and estimated power output of each was 11,000 e.h.p. at sea level and 2,300 e.h.p. at 36,000 ft.

The aircraft was 38 ft 19 in long and the wing span 26 ft 3 in. Performance was calculated as take-off and climb to 3,300 ft in 2 min, climb from 3,300 to 36,000 ft in 2·3 min, maximum speeds of 680 m.p.h. at sea level and 595 m.p.h. at 36,000 ft. At those speeds and altitudes the range was 145 miles and 435 miles respectively.

Heinkel planned a machine with a ramjet 16 ft long in each wing root, and the Skoda works in Czechoslovakia, while under German occupation, made studies of a design in which a 31 ft Saenger athodyd was mounted in an exceptionally bulky fuselage, with the pilot lying prone above. For a sea-level speed of 630 m.p.h. the thrust was 9,700 lb.

Delta-wing fighters by Lippisch were designed for propulsion by ramjets ; some to burn coal carried in a circular rotating basket in the duct. It was calculated that 1,760 lb of coal would give the Lippisch P.13 fighter a 45 min endurance.

Concerning British experiments very little can be said. A certain amount of small-scale wind tunnel work has been undertaken, and during 1946 a model of a hypothetical ramjet aircraft was displayed at Farnborough. In this design the pilot was to lie prone in a long enclosure above the duct, fuel was accommodated in the relatively thick wing, and the take-off was to be made under rocket power.

The Bristol company is actively engaged in the development of a ramjet missile from a design project initiated by the Royal Aircraft Establishment. Details are not yet available but a ' test vehicle ' was exhibited at Farnborough in 1952. This revealed a slender, tail-finned missile with two outrigged ramjets. Two solid-propellant booster rockets are used for launching and initial acceleration and then jettisoned. Maximum speed, it is suggested, is of the order of 1,500 m.p.h.

PULSEJETS

Early work on an intermittent impulse duct was undertaken by Karavodine in 1908. His idea was to furnish motive fluid for a turbine, and he used a long tube fitted with a unidirectional check valve at the forward end. After each explosion the inertia of the gas column created a zone of lowered pressure and effected the induction of fresh air past the check valve. The scheme was inefficient and an experimental model had a fuel consumption rate of about 5·5 lb/h.p./hr.

There followed a number of schemes in which a jet of combustion gases was ' augmented ' by air entrained by way of co-axial conical nozzles, after the manner of an ejector. In these the loss of velocity on mixing offset the gain in mass and the resulting advantage was trivial. Examples are shown in Chapter 2. In 1928 a German engineer, Schmidt, refuted the ' augmentation ' theory and advocated instead the direct transfer of energy from the combustion zone to the free air in a straight tube. This was the first conception of the intermittent impulse duct, or pulsejet. Schmidt was granted a patent in 1931 and continued his experimental work. The operational cycle is shown diagrammatically in

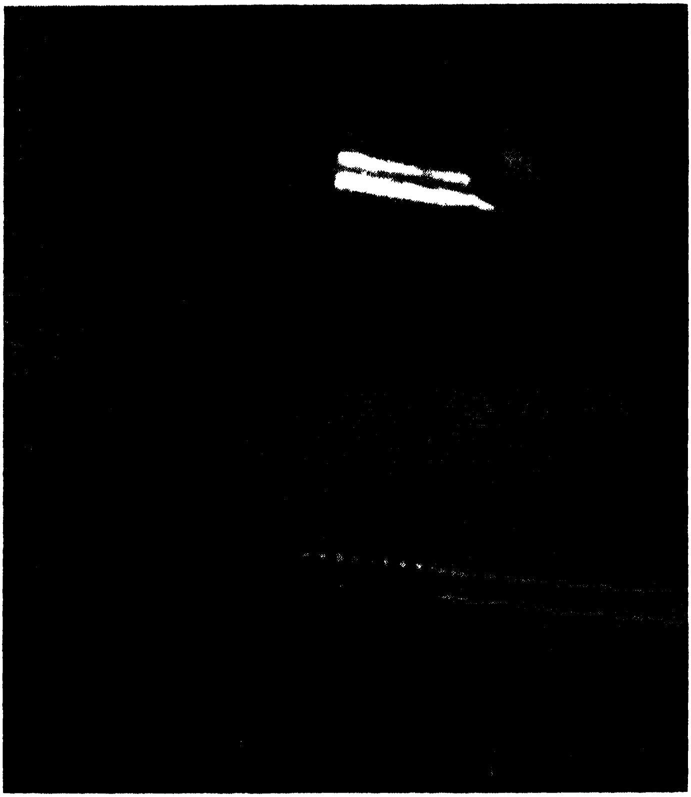


Fig. 14 : 9—Bristol ramjet test vehicle in operation. Upper: Rockets fall away at end of boost phase. Lower: Vehicle in flight under ramjet power.

Fig. 14 : 10, and the following description of the cycle is abstracted from Schmidt's memoirs :

' The main tube is provided with an inlet valve for combustible air. This valve, represented by a double line in the diagram, allows the fuel-air mixture to enter and prevents the return of the exhaust gas after the mixture is ignited, that is to say, during the generation of pressure inside the tube. At the top of the diagram the tube is shown at the end of a cycle ; near the inlet valve is a charge of fresh mixture. In the remainder of the tube there are burnt gases from the previous combustion, with a body of air near the outlet orifice. The second illustration shows the stage immediately after ignition. Small arrows indicate the effect of pressure due to the combustion ; this effect takes place very rapidly so that the gas column in the tube is pushed towards the rear, as shown in the third and fourth illustrations. Due to inertia, this

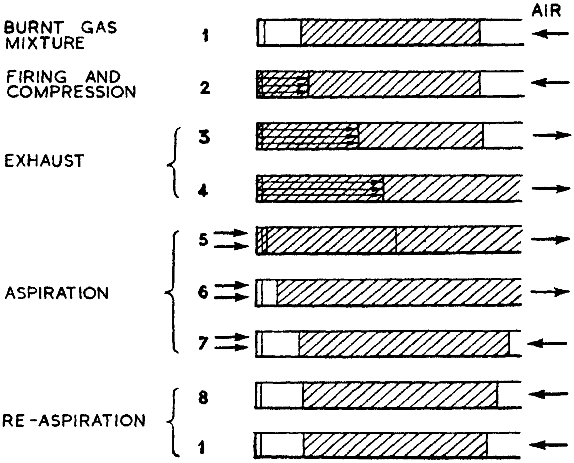


Fig. 14 : 10 – Cycle of operations in Schmidt-type pulsejet.

movement continues even after motive pressure has ceased. The depression which is then set up in the tube, particularly near the valve, draws a new column of mixture into the tube, as indicated by the arrows in stages 5, 6 and 7.

'Air also enters the tube through the exhaust outlet, the " re-aspiration " principle, and is forced out again in the next cycle. Thus the tube functions at pre-determined resonance frequency. Not only is there the automatic intake of fresh air but also automatic ignition at high combustion rates.'

The thermodynamic efficiency of a pulsejet is extremely low, but it is still able to retain its place as a power unit because its propulsive output is relatively high. This is due to the fact that each discharge is also followed by the induction of a certain volume of air into the exhaust end of the tail-pipe. This air is then expelled at the next successive combustion.

Thus the kinetic energy produced is applied to a larger mass of air which, although having the effect of lessening the exit-velocity, increases the momentum, and therefore the thrust, of the discharge gases. In 1942 the Argus Co. in Berlin was ordered to put it into production to power the Fi 103 missile, commonly known as the V-1 flying bomb. It was completely novel and both the aircraft and the power unit abounded in ingenious features.

The only moving parts in the pulsejet unit were the non-return valves in the frontal grid controlling the admission of air into the combustion chamber. Each of the 63 valves comprised a pair of rectangular flap plates anchored to vane plates at one end and lightly seating on each

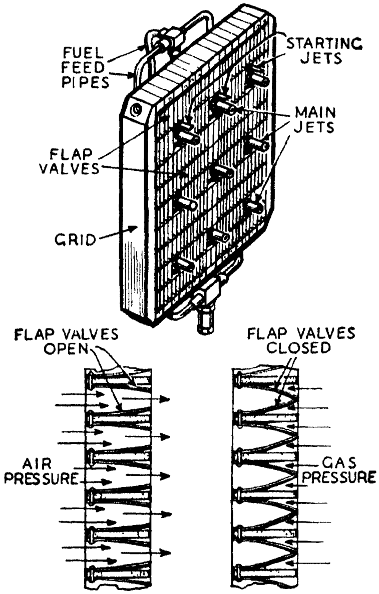


Fig 14 11—Diagram showing operation of V 1 check valves

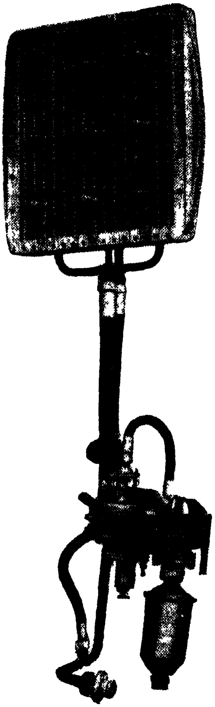


Fig 14 · 12 Air valve and fuel spray assembly

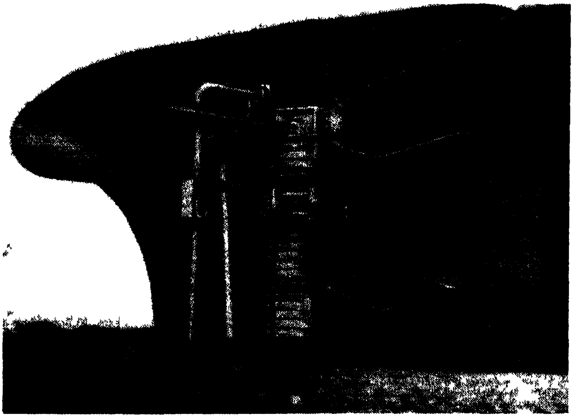


Fig. 14 : 13—Section of intake duct and valve grid, showing main and starting fuel sprayers.

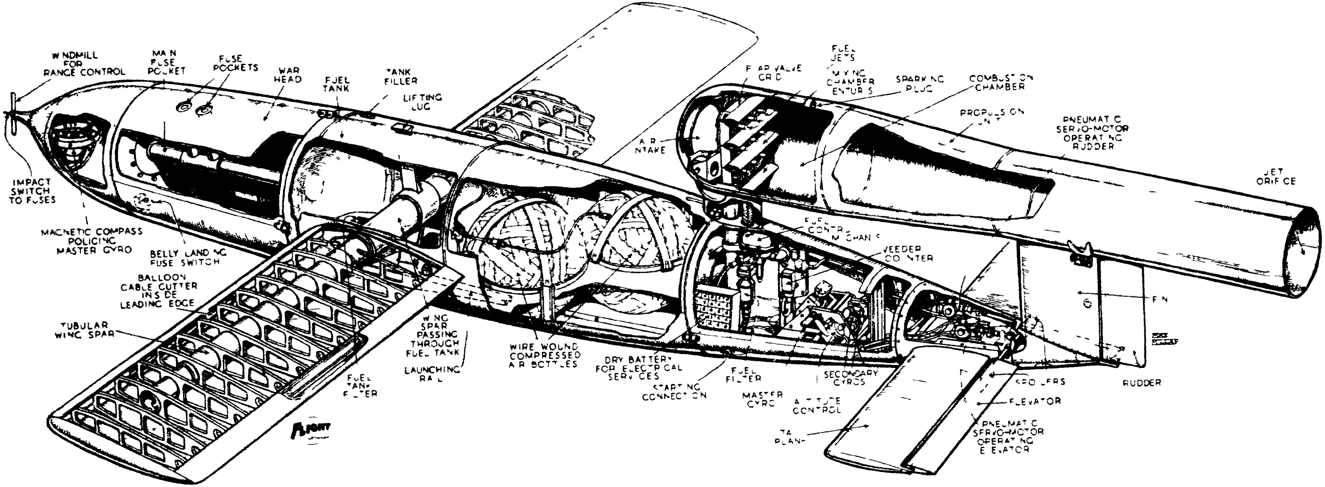


Fig. 14 : 14—The Fi 103 (V-1) flying bomb propelled by a pulsejet developing 600 lb thrust. A simple, parallel-wing aircraft built almost entirely of mild steel plate in five main components for easy transport and assembly. Wing spar a single 4·5 in-diameter solid-drawn steel tube threaded through the fuel tank. Air compressed to 150 atm and stored in two mild steel, wire-bound, 22 in-diameter spheres, furnished motive power for compass, autopilot and fuel supply. Warhead weighed about 2,000 lb or approximately 42 per cent of all-up weight at launch. Maximum speed about 360 m.p.h.

other at their inner edges. Two spring steel leaves formed each flap plate. Protruding through the grid were nine fuel sprayers arranged in three horizontal rows. Above the upper row were three nozzles through which compressed air from an external source could be supplied for starting the unit prior to launching. Initial ignition was by a glow plug located about 2 5 ft downstream from the grid but, once the duct was heated up, firing was automatic.

The duct was a welded shell of heavy-gauge mild steel having overall dimensions of 1 9 ft diameter and 11 25 ft long. Apart from the grid, the only internal fittings were four horizontal bars forming, in effect, venturis for the fuel jets. It was mounted above the rear of the fuselage on a three-point, rubber-jointed attachment. At the forward end a pressed-steel fork was trunnioned to the grid and at the rear it was secured to the fin.

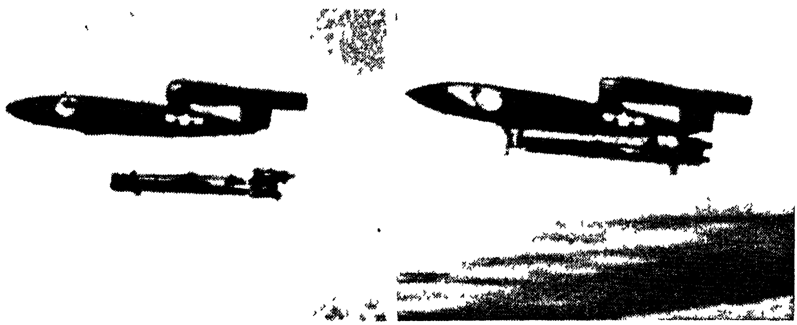


Fig. 14 : 15—U.S. flying bomb launched by rocket-propelled, jettisonable undercarriage.

The geometry of the duct gave operation at the rate of 45 pulses per sec or 2,700 per min. Although the unit would function statically the propulsive energy was insufficient to accelerate the missile. Accordingly, the missile was launched from a ramp by auxiliary thrust and rapidly accelerated to a speed of about 200 m.p.h.—that was 50 m.p.h. in excess of the stalling speed. Thereafter the unit functioned more efficiently, thrust would increase with the air-ramming effect, until eventually a velocity would be reached at which thrust and drag would balance and a stable flight speed would be established. This was reckoned to be 350–360 m.p.h. although modified versions, aerodynamically somewhat refined, were credited with an additional 50 m.p.h Ordinary motor spirit of relatively low octane rating was used as the fuel and the tankage was about 150 gal. Consumption was at the rate of about 1 gal/mile. As the thrust developed was 600 lb, at 360 m.p.h. the horse-power of an equivalent piston engine with an 80 per cent efficiency airscrew would be 720. Specific consumption was, therefore, at least eight times that of a piston engine. Doubtless this could be improved.

Main dimensions on performance characteristics of the missiles used to bombard London follow :

Overall length	25·375 ft
Fuselage length	21·5 ft
Fuselage diameter, max	2·7 ft
Power unit length	11·25 ft
Power unit diameter, max	1·9 ft
Wing span	17·67 ft
Wing chord	3·42 ft
Tail span	7·4 ft
All-up weight	4,750 lb
Wing loading	93·0 lb/ft^2
Warhead	2,000 lb
Flight speed	360 m p h.
Stalling speed	150 m p h
Thrust at 360 m p h	600 lb
T.h p at 360 m p h	576
Power loading	8·25 lb/t h p

Subsequent investigations indicated that the air valves rapidly lost their efficiency, the grid minimized ram effect, combustion could be improved and backflow of the gas column used more effectively.

In 1943 the Aerojet Corp. in America experimented with an ‘aero-pulse’ duct and by early 1944 the U.S.N. had developed a ‘resojet’ with no mechanical valve arrangement but instead an attenuating tube —a so-called acoustical valve. Specimens of the German V-1 unit were retrieved during the summer of 1944 and thereafter a number of studies and experiments were initiated. In December of that year the Ford

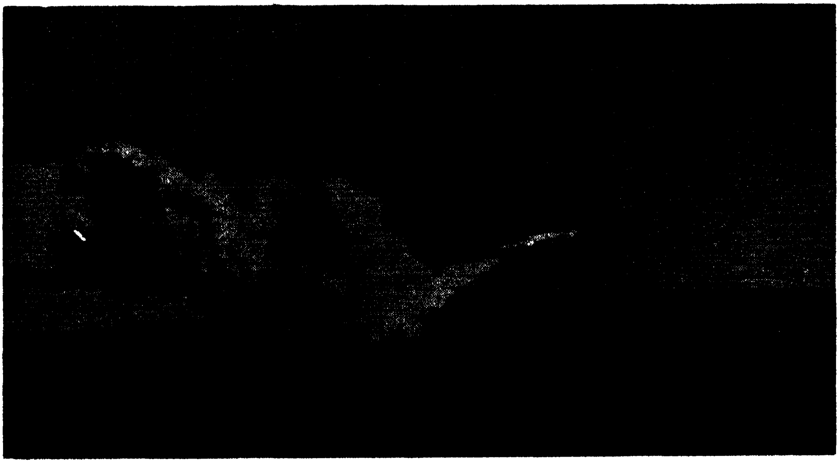


Fig. 14 : 16—Loon pulsejet missile rocket-launched from U.S.N. submarine.

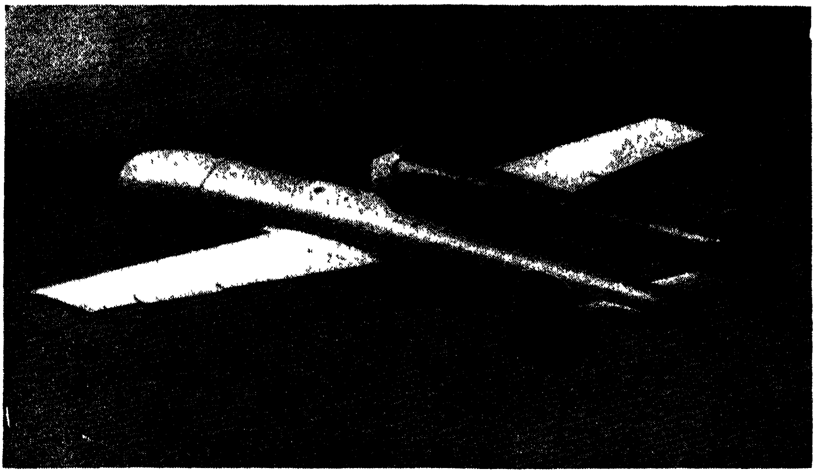


Fig. 14 : 17 U.S.N. remotely controlled gunnery target aircraft powered by pulsejet.

Motor Co. were conducting static tests of a pulsejet which had been produced in an amazingly short period to a design based on the V-1 unit. Two months later these were flight-tested. They were launched from a ramp by means of a rocket-propelled undercarriage which was automatically jettisoned when the missile was operationally airborne.

The U.S. Navy sponsored its development and demonstrated the practicability of launching such missiles from both warships and submarines. One of these missiles, the Loon, which was reported to fly at a speed of between 400 and 500 m.p.h., is shown being launched in 1949 from the U.S. submarine Carbonero off Hawaii.

Another version, designated KD2G-2, had a twin-finned tail and was used as a pilotless gunnery target. It was remotely controlled from air, land or sea and made a descent by parachute for possible recovery and re-use.

France also built a radio-controlled target aircraft, the Arsenal 5.501, based on the design of the German Fi 103. First displayed in Paris in 1951, it was arranged for launching by a 39 ft-long catapult which gave it an initial speed of 240 m.p.h. Descent was by parachute and approaching the ground a proximity fuse fired two braking rockets to reduce landing shock. Later it was adapted for a rocket-assisted launch from a portable ramp mounted on a lorry.

A valveless pulsejet, the Escopette, has been developed by the S.N.E.C.M.A. organization in France. In multiple installations of four and six units they have been used to power the Emouchet sailplane and were first flown publicly at the display in conjunction with the 1951

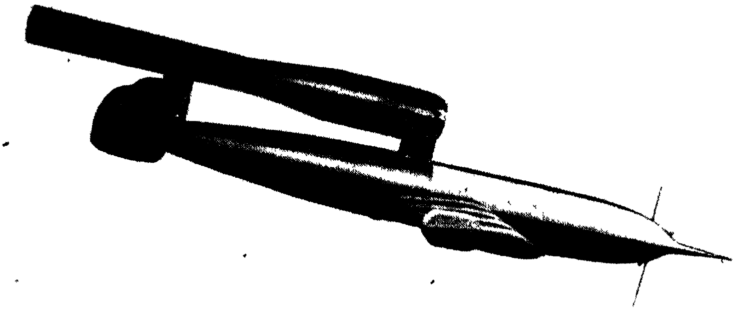


Fig. 14 : 18—French pulsejet target aircraft, the Arsenal 5.501.

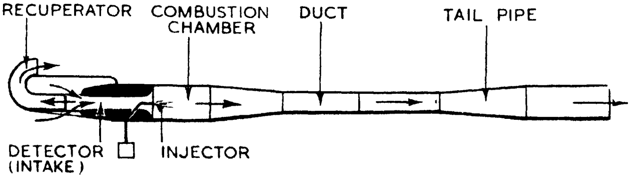


Fig. 14 : 19—Diagrammatic section of Escopette pulsejet.

Fig. 14 : 20—Group of three Escopette pulsejets installed under wing of Emouchet sailplane.

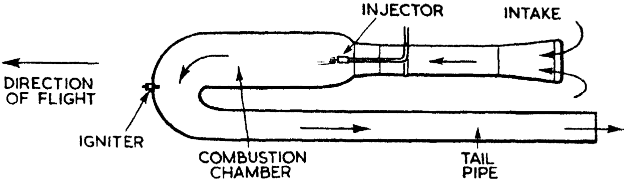


Fig. 14 : 21– S.N.E.C.M.A. Ecrevisse pulsejet.

Paris show. Shown diagrammatically in Fig. 14 : 19, the unit comprises a so-called detector or choke, combustion chamber, duct and tail pipe.

The detector, in the form of a convergent-divergent nozzle, accelerates and then decelerates the intake air ready for combustion of the fuel which is sprayed downstream from an injector located at the mouth of the combustion chamber. Naturally, the detector cannot function as a uni-directional valve and during each cycle a considerable reverse flow is produced. This apparent disadvantage, however, is turned to useful effect by a recuperator tube which turns the reverse flow through 180° to discharge rearwardly and provide additional thrust. To facilitate the admission of air to the unit, the recuperator is spaced from the mouth of the detector ; the gap may be discerned in Fig. 14 : 20 showing three units grouped below the wing of the sailplane.

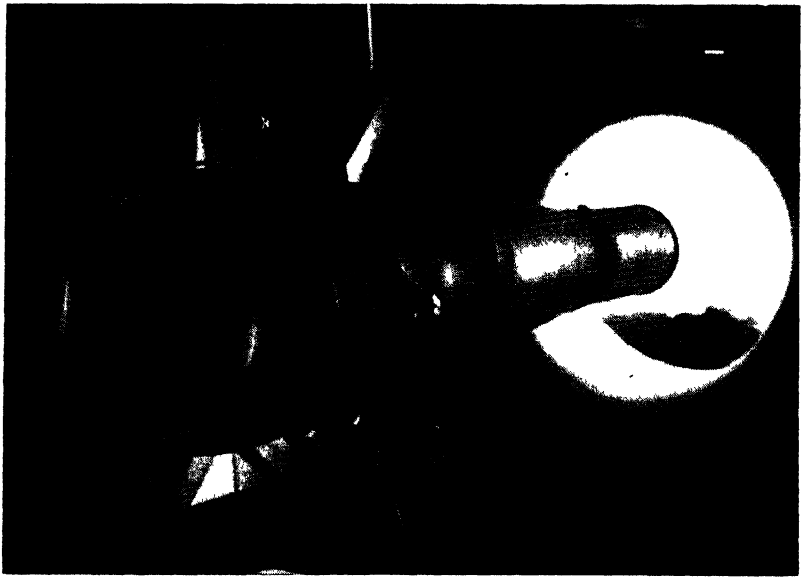


Fig. 14 : 22—Saunders-Roe pulsejet on test.

Petrol of any octane rating is used as fuel and is fed from a tank by compressed air at a pressure of 4·4 to 20·6 lb/in^2. According to the type of sprayers fitted the thrust developed ranges from 6·6 to 22·0 lb with a fuel consumption of 0·24 to 0·66 lb/min. The installed weight is 11·24 lb per unit.

On the ground a start is effected by injections of air at a pressure of 44 to 74 lb/in^2 from a bottle by means of a small nozzle inserted in the mouth of the detector intake. Electric ignition is required only for the initial start. In the air, starting or restarting presents no difficulty at speeds in excess of 45 m.p.h. The maximum duration of a four-unit installation with a 4·4 gal tank on the Emouchet sailplane was 10 min at full throttle at ground level.

In a later version, the Ecrevisse, the recuperator tube is dispensed with and the duct is formed with a return bend so that both the intake and the propulsion nozzle are directed rearwards. This unit is 97·5 in long and weighs only 22 lb. Static thrust developed is 66·2 lb at a fuel consumption rate of 1·2 lb/hr/lb. Versions of the Ecrevisse have been used to power ‘ tail-standing ’ vertical-take-off prototype aircraft.

In England, Saunders-Roe, Ltd. have developed two pulsejets for helicopters, guided missiles or sailplanes. The smaller unit, first exhibited in August 1953, is 47·5 in long, 5·5 in maximum diameter and weighs 15·5 lb. Normally it is operated at 120 c/s and develops 45 lb thrust. Variation of the chamber design enables frequencies of from 45 c/s to 200 c/s to be obtained. The larger pulsejet, shown running in the test cell, is rated to produce 120 lb thrust.

ROCKETS

In the development and application of rocket propulsion for aircraft and missiles Germany showed the way in the years up to 1945. When authority was assumed by the Nazi party in the early 1930s, rocket research was accelerated under conditions of strictest secrecy. Several new propellants were brought into practical production and operation. Few authorities outside that country could have foreseen in 1940 that in a space of five years rocket-powered interceptor fighters would be attacking Allied bombers and that massive war rockets would be bombarding London from the other side of the Channel.

By exploiting captured German material, adapted and prepared with the aid of German technicians, America and other powers have in post-war years achieved remarkable success, especially in respect of guided missiles. An American-built liquid rocket unit enabled the Bell X-1 research aircraft to exceed the speed of sound at a time when the majority of experts considered controlled supersonic flight to be impossible of attainment for months or, if truth be told, years ahead. It must, however, be recorded in this connection that designs were prepared during the war for a British research machine, the Miles M.52, to achieve supersonic speed with a turbojet and exhaust reheat.

For air-launched or ground-launched projectiles and for assisting the take-off of aircraft, solid-fuel and liquid-fuel rockets have been extensively employed. Aircraft and the larger types of missiles generally have relatively elaborate liquid-fuel units.

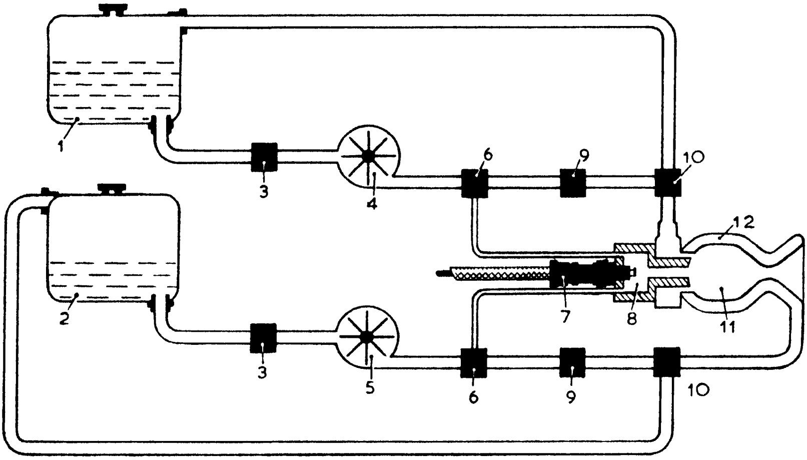
As observed by Mr. Thomas F. Reinhardt, of the Bell Aircraft Corporation, solid propellants may show up somewhat more favourably than liquids inasmuch as the space saved by eliminating piping, valves, and pumping system can be used to store a larger volume of propellants. The principal objection to a solid propellant is that there is no opportunity for control ; i.e., after the propellant is ignited it must continue to burn until all is consumed. Liquid-propellant rockets may be started and stopped at will. Another serious objection to the solid propellant is the difficulty in controlling the centre of gravity during burning. This is complicated by the fact that the design of a solid-propellant charge to obtain a definite rate of burning is very difficult, even without the added restriction of maintaining c.g. control. Solid rockets have no provision for cooling, and depend upon the heat capacity of the metal walls to prevent overheating. For burning times much in excess of 20 sec the mass of metal required becomes prohibitive. Thus the flexibility of control of liquid propellant rockets makes them the logical choice for man-carrying aircraft.

After a number of abortive demonstrations with Sander's solid-fuel rockets on road and rail vehicles in Germany, Fritz von Opel, the motor car manufacturer, in 1928 commenced experiments in aircraft propulsion. On September 30, 1929, he achieved the first successful rocket-powered flight.

Credit for the production of the first liquid-fuel rocket motor goes to Dr. R. H. Goddard, the American scientist who ran a liquid-oxygen/petrol unit on the test bed on November 1, 1923, and launched the first liquid-fuel rocket on March 16, 1926, at Auburn, Massachusetts. It had a burning time of 2·5 sec and travelled, in a high trajectory, a distance of 184 ft at an average speed of 60 m.p.h.

The operation of a bi-propellant rocket motor can be followed from the schematic layout in Fig. 14 : 23. Control from the cockpit is by a master switch to start or stop the motor and a control switch to select either part thrust or full thrust. The unit can be started and stopped as frequently as desired and changes of thrust in response to control are virtually instantaneous.

Operation of the master switch starts the rotation of oxidant and fuel pumps, both pumps are primed and igniter valves are opened. As the two propellants flow to the small igniter chamber they are fired by the



1. Oxidant tank : 2. Fuel tank : 3. Suction valves : 4. Oxidant pump : 5. Fuel pump : 6. Igniter valves : 7. Spark plug : 8. Igniter chamber : 9. Throttle valves : 10. Stop valves : 11. Combustion chamber : 12. Cooling jacket.

Fig. 14 : 23—Operational diagram of bi-propellant rocket motor.

spark plug and a pencil of flame is projected into the combustion chamber. The main stop valves then open and the by-pass valves, allowing propellants to flow back to the tanks, close. Oxidant flows directly into the combustion chamber but the fuel first passes through the chamber jacket to exercise a cooling function. In the chamber the propellants are ignited by the pencil of flame, the motor fires, ignition current is cut off and igniter valves are closed.

The amount of thrust developed can be pre-selected before operating

Propellants	*Volume ratio*	*Combustion Temperature, °K*	*Specific impulse, lb/lb/sec*
Liquid hydrogen and liquid oxygen ..	1 : 3·2	2,560	343
Alcohol and liquid oxygen	1 : 1·15	2,780	235
Gasoline and liquid oxygen	1 : 1·4	2,240	225
Amine compounds (R–Stoff) and nitric acid (SV–Stoff)		2,290	200
Aniline and nitric acid			187
Hydrazine hydrate (C–Stoff) and hydrogen peroxide (T–Stoff) ..	1 : 2·03	2,080	180

C–Stoff: Hydrazine hydrate 30 per cent, methanol 57 per cent, and water 13 per cent.
T–Stoff: Hydrogen peroxide 80 per cent and water 20 per cent.
R–Stoff: Crude oxide m-xylidine 57 per cent and tri-ethylamine 43 per cent.
SV–Stoff: Fuming nitric acid 90 to 98 per cent and remainder sulphuric acid.

the master switch and the control may be either a multi-positional or smoothly variable type. Safety devices ensure that starting operations occur in the correct sequence. The mal-functioning of any component leads automatically to a shutdown.

The diagram indicates centrifugal pumps to feed fuel and oxidizer to the combustion chamber. These require driving from an external source of power. Usually this takes the form of a turbine fed by gases generated from the main propellants or from other suitable propellants. Where such propellants yield a gas of too high a temperature a third liquid may be injected into the gas generator to act as a diluent and lower the temperature to a safe value.

Alternatively, the propellants may be fed from tanks pressurized by an inert gas, such as nitrogen or helium, carried in storage bottles. While being relatively simple, this system is both bulky and heavy since all tanks must be strongly constructed to withstand the pressures. It thus tends to be unsuitable for large-range rockets or for installation in turbojet aircraft for power boosting.

CHAPTER 15

Rotating Wing Propulsion

VERTICAL ascent by means of a rotating horizontal propeller has been known from the earliest times. The origin of the Chinese helicopter toy, in which a propeller is thrust up a screwed shaft to make a short ascent and then windmill down to the ground, is lost in antiquity. Leonardo da Vinci, at the turn of the 15th century, made notes and drawings of a helicopter with which he is reputed to have experimented. The principles advanced were basically sound but at that period of history, of course, he lacked a source of mechanical power. More than 350 years after da Vinci's death designs were still being produced for manually-driven helicopters. By that time, however, the need for mechanical propulsion was realized and models were constructed with compressed-air or steam power units. During the period 1885–1895 the American scientist T. A. Edison experimented with electrically-driven models, and Sir Charles Parsons, the British steam turbine pioneer, built and flew a steam-driven model.

Quite early the advantages offered by reaction-propelled rotors were recognized and there are records of a steam-jet propelled model built by W. H. Phillips making a flight at Primrose Hill, London, in 1842. Jet-propelled rotors in the modern sense may be said to date from a French patent granted to M. A. Quentin in 1909 for a helicopter having four twin-bladed rotors. In this a compressor supplies air to a mixing chamber where fuel is added and the resulting combustible mixture is fed by way of the rotor shafts to combustion chambers in the blades.

Since that time the Patent records reveal that the subject has continuously engaged the attention of inventors in Britain, U.S.A., France and Germany. Scores of designs including ramjet, pulsejet and simple jet types, practical and impractical, over-elaborated and over-simplified, have appeared. Despite the distraction of their variety, in total they represent an accumulation of investigation that well repays study.

Some early proposals now appear quaint or fantastic but even so should not be casually dismissed. The 1912 notion of two Frenchmen for a jet-driven single-bladed rotor balanced by a stub blade housing an engine driving a blower (Fig. 15 : 1) reappeared in 1945 in a patent of the United Aircraft Corp., U.S.A., with a free-piston gas generator in the stub.

Many early designs fed fuel under pressure to tip-mounted jet units but depended upon centrifugal action to compress the air supply taken in

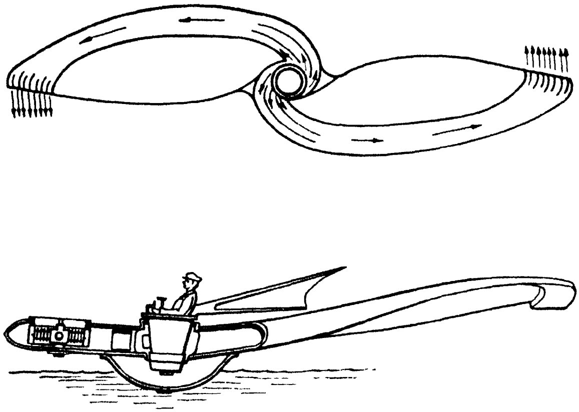


Fig. 15 : 1 —The single-bladed rotor of Papin and Rouilly and, above, a two-bladed alternative.

near the rotor hub. Another proposal which enjoyed some popularity was for jet-driven propellers mounted on the blade tips. Then followed longitudinally slotted blades from which the efflux was discharged to assist lift or to energize the boundary layer. A Sikorsky scheme of 1929 with leading-edge and trailing-edge slots is shown in Fig. 15 : 2.

In some schemes the propulsion rotor was an independent component running at high speed and driving the sustaining rotor or rotors through a reduction gear. G. and J. Weir in Britain in 1942 and A. Gazda, U.S.A., in 1943, suggested the use of compressed air, engine cooling air, and exhaust gases discharged from a lateral nozzle in the tail of the fuselage for torque compensation on mechanically driven helicopters.

The following year L. H. Leonard obtained two U.S. patents for jet-driven rotating-wing aircraft intended for vertical ascent and subsequent change of attitude to the horizontal for level flight. In one the blades continue to rotate in horizontal flight whilst in the other the blades were held stationary to serve as fixed wings. It was at that period the German Focke-Wulf concern designed the remarkable rotating-wing fighter (Fig.

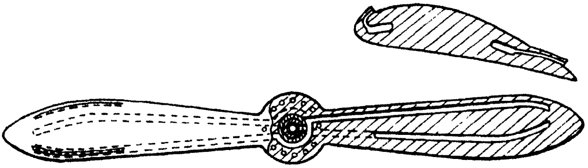


Fig. 15 : 2—A slotted blade from a 1929 Sikorsky patent.

15 : 3). In place of the normal wings were three rotor vanes of adjustable pitch. Each vane was furnished at the tip with a ramjet unit and also a Walter auxiliary rocket unit for starting up and for emergency landing in the event of the ramjets cutting out. In operation, the helical velocity of the ramjet units would be considerably higher than the linear speed of the fuselage.

For take-off the aircraft stood vertically on its tail undercarriage and, with the vanes in neutral pitch, initial rotation was effected by the auxiliary rockets. When an adequate velocity had been attained the ramjets would be started up. On the vanes being moved into fine pitch vertical lift would be imparted, due partly to rotation of the vanes and partly to components of ramjet thrust parallel to the fuselage axis. When the machine was levelled out the pitch of the vanes would be further coarsened and the rotational speed reduced to maintain the air velocity at the ramjets at a constant Mach No. of 0·9.

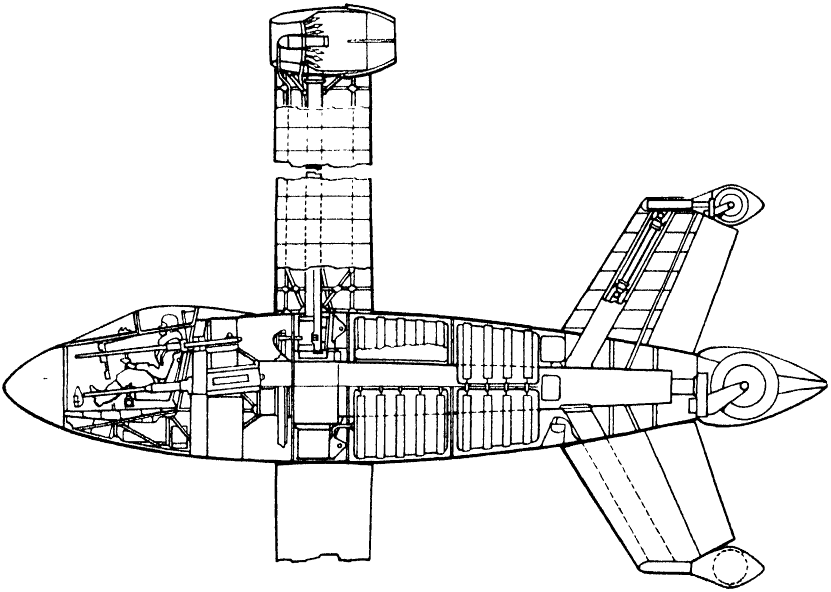


Fig. 15 : 3—Focke-Wulf ramjet-propelled rotating-wing fighter project.

The specification gave the following characteristics :

Fuselage length	30 ft 8 in
Rotor diameter	37 ft 5 in
Ramjets, length	4 ft 7 in
Ramjets, diameter	2 ft 3 in
Fuel, ramjets	3,300 lb
Fuel, rockets	200 lb
Take-off weight	11,400 lb

Anticipated performance was :

Speed, zero altitude	620 m.p.h.
Speed, 36,000 ft	520 m.p.h.
Cruise at 385 mp.h. at 36,000 ft	1,500 miles 3·7 hr
Climb to 36,000 ft	3·5 min 507 lb fuel

Complete turbine power units attached to the rotor blade tips figure in a Fairey patent of 1944. Both turbojet and turboprop versions are shown (Fig. 15 : 4) and common features are an axial compressor, annular combustion chamber and a cooling air duct around the centre shaft and bearings of the rotating assembly. Starting is effected by an electric motor in the central casing forward of the compressor and fuel ignition lines are carried through the blade main spar. The turbine of the jet unit has, as usual, only a single stage, but that for the turboprop has three stages.

While the mounting of turbine units at the blade tips avoids the difficulties and losses involved in ducting the heated, low-density efflux of a fuselage-mounted turbine through the rotor hub and blades to the tips, it presents its own problems. The undesirable concentration of weight at the blade tips introduces difficulties arising from gyroscopic reaction and blade inertia. To circumvent such disadvantages the Fairey company in 1947 patented the Doblhoff low-pressure system shown diagrammatically in Fig. 15 : 5. A turbine-driven centrifugal compressor delivers air at a relatively low pressure through a heat exchanger to the rotor hub whence it is ducted along the blade to the propulsion nozzles at the tips. Air for the turbine is tapped off the duct leaving the heat exchanger and heated in a combustion chamber. The turbine efflux passes through the heat exchanger and may either be directed to flow over a rudder or conducted to lateral nozzles at the tail.

In order that the cross-section of the rotor blades is not raised above the aerodynamic optimum a plurality of air conduits of small cross-sectional area may be used. The area of the jet orifice may be automatically adjusted by means of shutters controlled by a centrifugal device influenced by the angular velocity of the blade tip and a pitot tube and servo cylinder influenced by air density.

Another automatic control, responsive to compressor delivery pressure, operates to fine the pitch of the rotor blades for autorotation in the event of a failure of either the turbine or compressor.

Since the termination of the Second World War a tremendous development of the helicopter has occurred and nowhere has activity in this sphere been so intensive as in the U.S.A. Helicopters have proved their utility and reliability and are now used operationally on military and naval duties. While the majority employ a piston engine and mechanical drive for the rotors, numerous examples with jet-propelled rotors have

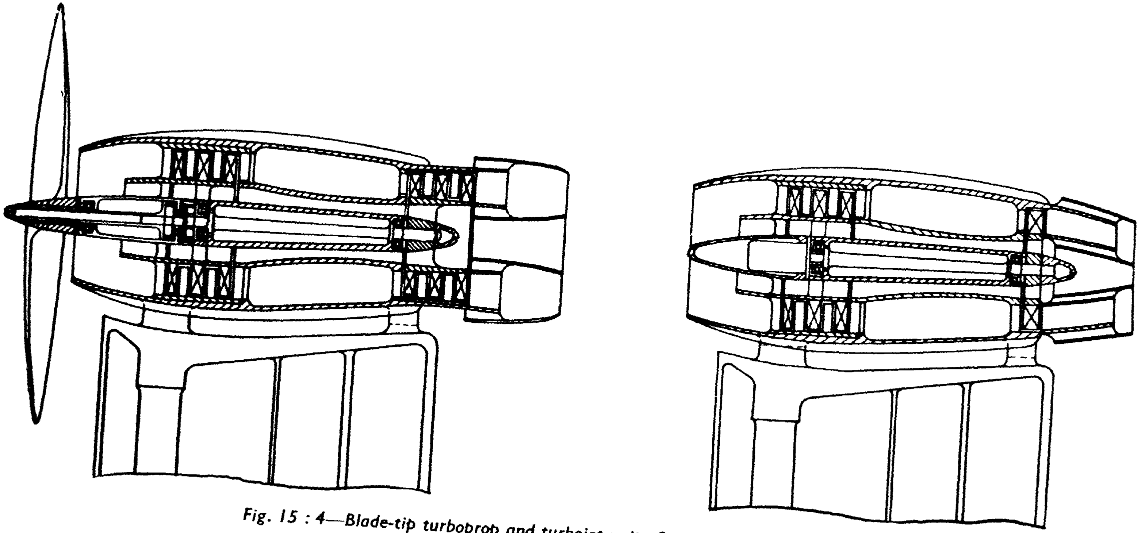


Fig. 15 : 4—Blade-tip turboprop and turbojet units for a 1944 Fairey scheme.

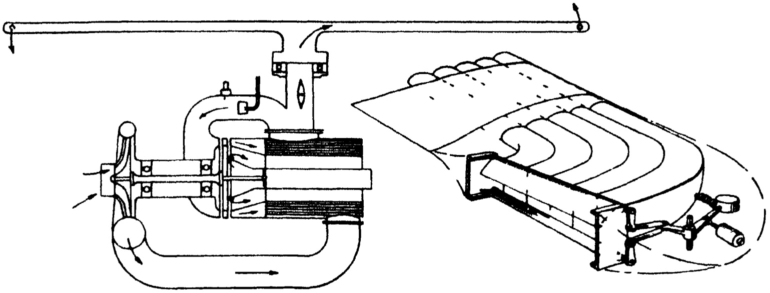


Fig. 15 : 5—The Doblhoff low-pressure jet rotation system.

been built and flown. With the advent of the large helicopter, for both civil and military operation, there would seem little doubt that jet-propulsion in one form or another will become more general. Indeed, no less an authority than I. I. Sikorsky, the pioneer designer of the most widely used helicopter, has stated : ‘ Giant machines, in all probability, will not be designed as enlarged copies of a modern small helicopter. In particular, in very large helicopters it would be more practical to apply the power directly to the tips of the blades by the use of jets, small auxiliary propellers, or other devices ’

Two American firms have built and flown experimental pulsejet helicopters. The Marquardt M-14, Whirlajet, had a 29 ft-diameter two-

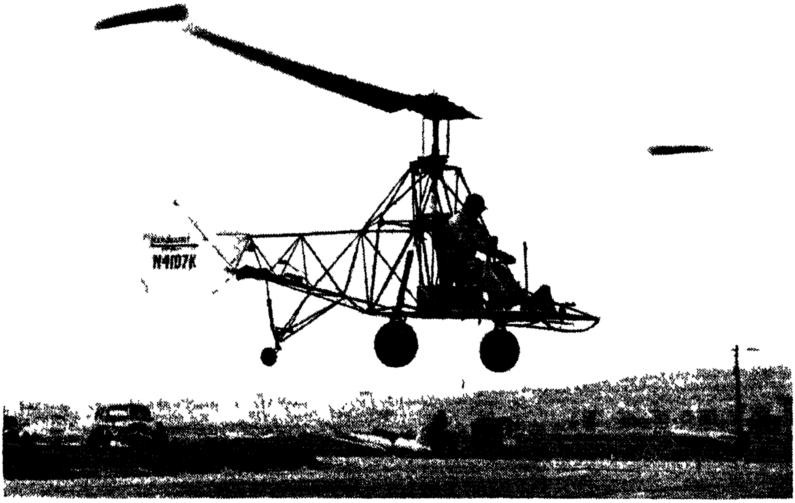


Fig. 15 : 6—Marquardt M-14 pulsejet helicopter.

bladed rotor with 8 in-diameter tip-mounted pulsejets supplied with fuel fed through the hollow blades. For the first test flights in 1948 starting was effected by directing a jet of air from a portable compressor into the intakes. After ignition and the establishment of resonance the air compressor was removed and the rotor freed for power operation. Self-contained starting equipment, comprising a compressor and storage tanks in the fuselage and conduits in the rotor blades, was to be installed subsequently. The source of ignition was a coil and battery which passed current from slip rings on the rotor hub. Weight was about 1,000 lb.

The American Helicopter Co. in 1949 conducted ground and flight tests of a two-seater helicopter designated XA-5, Top Sergeant. Powered by two pulsejet units it had a 33 ft-diameter rotor and weighed about 900 lb when empty. Each power unit weighed only 25 lb and had only one moving component, the valve block. In 1950 it was reported that these had a working life of 30 hr and could be replaced in the field in 5 min. Later the company produced a single-seater model, XA-6, Buck Private, having an open airframe of aluminium tubing and weighing when empty only 200 lb. This was tanked for an endurance of 90 min in the air.

In 1951 the company won a design competition and received a U.S.A.F. contract to build the XH-26. First flight tests were made 12 months later. For military purposes it is designed as a single-seater to carry twice its own empty weight of 300 lb. It is collapsible, is small enough to be carried aboard larger aircraft, and can be dropped by parachute in a container 5 ft × 5 ft × 14 ft. With only the simplest standard hand tools it can be reassembled in the field by two men in 20 min. Either a three-wheeled undercarriage or twin skids may be fitted.

Starting, for individual jet units, is by push button on the control stick. Air and fuel is injected and ignition is by a spark plug which is

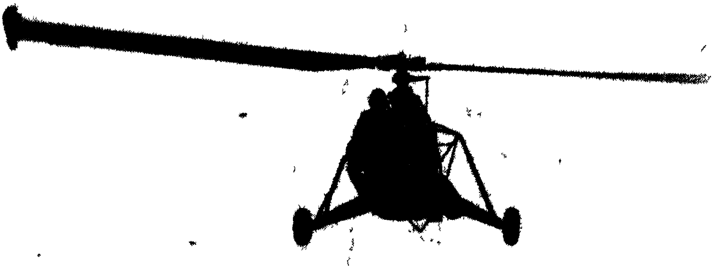


Fig. 15 : 7—American Helicopter Co.'s experimental pulsejet helicopter XA-5 Top Sergeant.

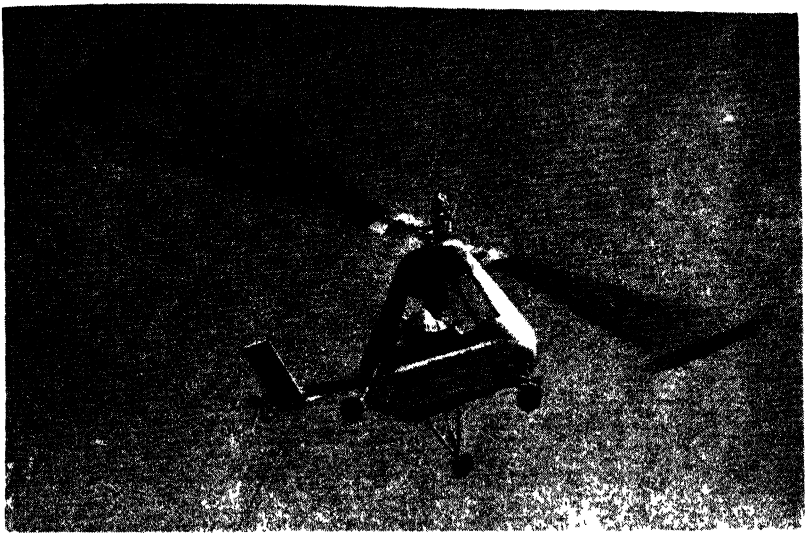


Fig. 15 8 American Helicopter XH-26 single-seat pulsejet helicopter for the U.S Army.

cut out as soon as firing has been established. Petrol, kerosine or diesel oil may be used as fuel and no warming-up is required. Thirty seconds is the stipulated period to start-up and become airborne. In these more highly developed power units the life of a valve block is rated at 50 hr.

The jet units are freely swivelled to the rotor blade tips and automatically assume the horizontal position in operation. Change of the blade pitch does not affect the position of the jet units. Differing from the earlier models, a small belt-driven tail rotor is fitted. It is, of course, not required as an anti-torque device but is provided to improve directional control and facilitate precise manœuvring. With an endurance of 90 min the XH-26 can ascend to altitudes comparable with those attained by conventionally powered craft. Maximum speed in level flight is 80 m.p.h

The ramjet has also received attention in America as a propulsion unit for helicopter rotors. In 1947 the McDonnell Corp. built the experimental J-1 Little Henry, later XH-20, helicopter. It was developed for the U.S.A.F. solely as a flying test bed for the propulsive system and did not have immediate operational application. The tip-mounted, stainless-steel ramjet units weigh only 10 lb each and have a peripheral velocity of about 600 ft/sec or 410 m.p.h. Propane is used as fuel and located behind the pilot is a combustion heater to preheat the fuel before it is fed to the power units. An auxiliary starter is fitted to speed the rotor initially until the tip velocity is sufficient for the ramjets to function.

In early flights in 1948 a speed of 50 m.p.h. was attained but doubt-

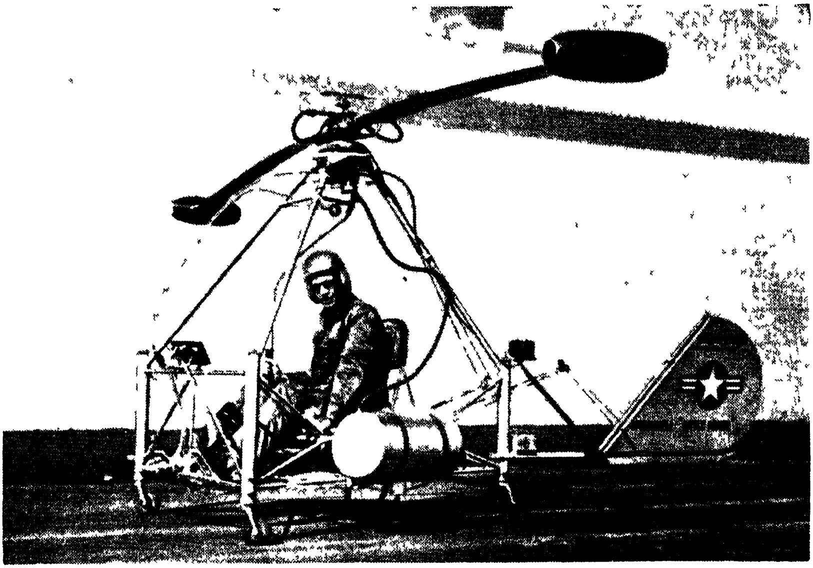


Fig. 15 . 9—McDonnell XH-20 ramjet helicopter.

less this figure has since been exceeded. Development is being continued and will include conversion to operation on standard aviation fuel. Main dimensions are : length 12 ft 6 in, height 7 ft ; the weight is 310 lb Another McDonnell helicopter, intended as a 'flying crane' for the U.S. Marine Corps, is expected to have a single large-diameter rotor with tip-jet drive.

In 1950 it was announced that the General Electric Co. was actively studying and experimenting with a ramjet for light, civil-type helicopters. They were developing an aluminium ramjet, designated 3R-1, which weighed only 5 lb.

Although defence requirements necessitated a concentration on military and naval helicopters and postponed production of the small, inexpensive Hiller HJ-2 Hornet, development is being continued. This interesting two-seater has been flown as an open type, on wheels or skids, and also completely enclosed as shown in Fig. 15 : 10. Propulsion is by two ramjets each weighing 12 lb and developing about 35 lb thrust. Attached by two bolts to the rotor tips, they can be removed in 3 min.

The 23 ft-diameter rotor, which must be driven at 150 r.p.m. before the ramjets can function, normally runs at 500 r.p.m. giving a tip speed of about 450 m.p.h. Initial rotation may be effected by an electric motor and battery or by a small petrol engine, hand-started by a cord. No warming up is required and a take-off can be made in 90 sec. Rate of climb is initially 1,100 ft/min and the estimated ceiling is 12,000 ft.

Cruising and maximum forward speeds are 70 and 85 m.p.h. respectively. The empty weight is 340 lb and with a gross weight of 900 lb the range is 50 miles and endurance 40 min on 30 gal of fuel. With a single occupant the range is about 100 miles. Hiller also have a project for a 40–50 passenger helicopter powered by ramjets.

Probably the simplest method of applying turbine power to helicopters is to substitute a turbine for the piston engine employed to drive the rotor through a mechanical transmission. This has been done experimentally on the Kaman K-225, a three-seater that normally is powered with a 225 h.p. Lycoming engine. It embodies the closely pitched, counter-rotating, intermeshing rotors which are the characteristic feature of Kaman designs. The turbine unit is the Boeing 502 (Fig. 16 : 19), developing 175 h.p. at the output shaft. Such an installation offers the advantages of reduced weight, cheaper fuel and less vibration, but at the expense of a higher rate of fuel consumption.

Piasecki has developed a large twin-engined, tandem-rotor transport and rescue helicopter, XH-16, for the U.S A.F. The first prototype was powered by Pratt and Whitney Double Wasp engines but the second is fitted with a pair of specially designed variants of the Allison T38 turboprop, each developing rather more than 2,500 h p.

Rotor propulsion by blade-tip air jets is adopted for the French S.O.

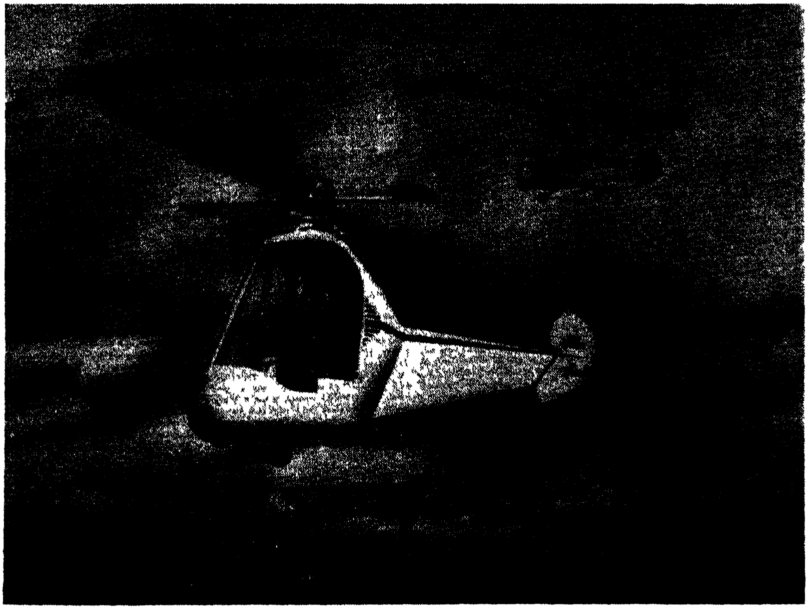


Fig. 15 : 10—The Hiller Hornet two-seat ramjet helicopter is used as a trainer.

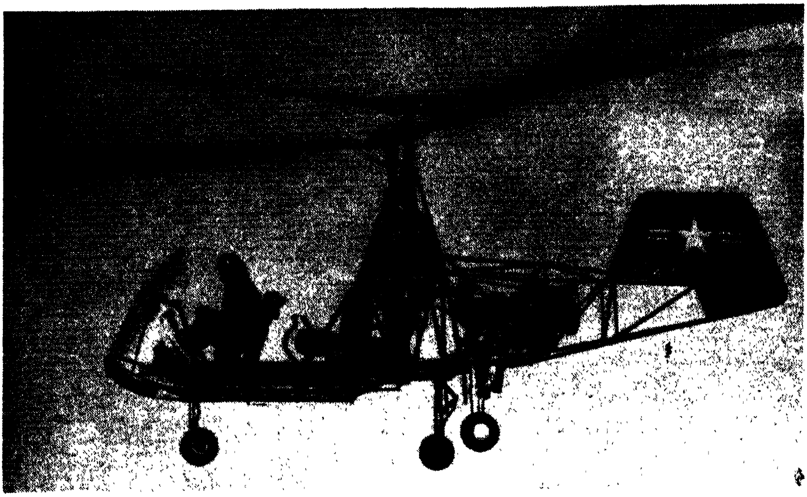


Fig. 15 : 11—Kaman K-225 three-seater with Boeing turbine and mechanical drive.

1220 Djinn single-seater. The power unit is the Turboméca Palouste turbine-compressor unit (Fig. 12 : 15) which delivers air at 41·2 lb/in^2 pressure at the rate of 139 lb/min. The Djinn is stated to be capable of carrying a useful load equal to its own weight. The normal gross weight is 1,390 lb but a vertical take-off can be made with a maximum of 1,580 lb. Maximum and cruising speeds are 81 m.p.h. and 62 m.p.h. respectively. The Djinn has now been built in some numbers.

Turbine units are also used to supply air to blade-tip combustion chambers. The largest helicopter yet built, the Hughes XH-17, first flown in California in October 1952, employs this system of propulsion.

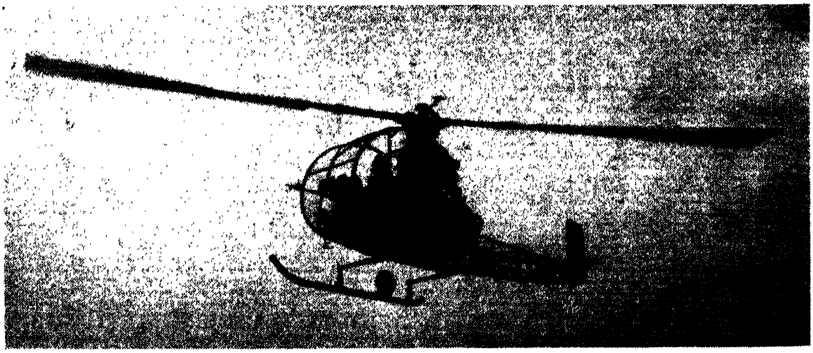


Fig. 15 : 12—S.O. 1220 Djinn helicopter with air jet propulsion.

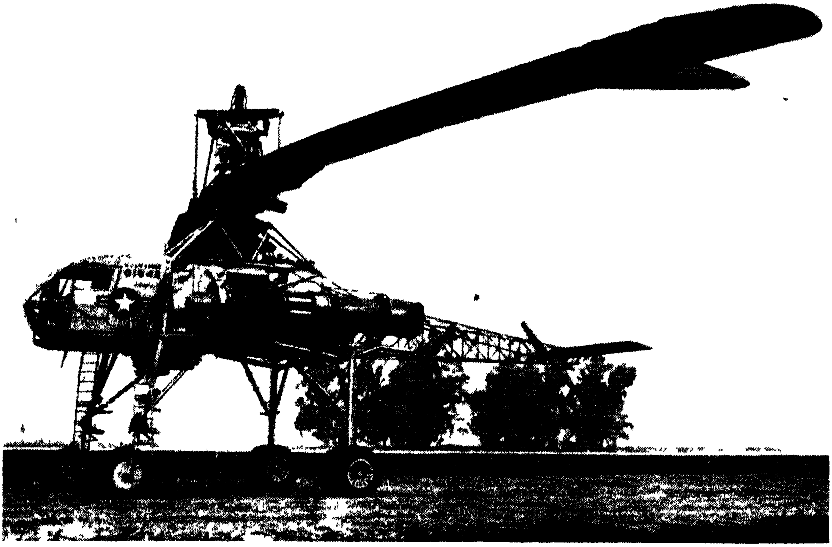


Fig. 15 : 13—Hughes XH-17 cargo-carrying helicopter has a 130 ft-diameter, jet-propelled rotor.

The two-bladed rotor is 130 ft-diameter and the overall height exceeds 30 ft. It is an experimental heavy-lift machine and may be regarded as a prototype of future heavy cargo carriers and so-called ‘ flying-cranes.’ The giant undercarriage is intended to straddle such loads as tanks, guns, lorries, bridge sections or machinery, for direct lift and transport to or from otherwise inaccessible areas.

The two power units, mounted externally one on each side of the fuselage in this test model, are modified General Electric J35 turbojets. Air for propulsion is tapped off the compressors of these units and fed through the rotor hub and blades to multiple jet tip burners, also of General Electric design. It will be noted that a small, mechanically driven tail rotor is provided to facilitate manœuvring. The range is 40 miles with a maximum speed of 80 m.p.h. and the ceiling height is reported to be 15,000 ft. Gross weight is 52,000 lb. Studies have been made for a civil project in which the power units are mounted above a capacious, double-decked fuselage.

Such combustion jets figure on another helicopter produced in France by the Société Nationale de Constructions Aéronautiques du Sud-Ouest (S.N.C.A.S.O.), the three-seater S.O. 1120 Ariel III. For this a Turboméca Arrius I turbine-compressor unit, with an air delivery of 167 lb/min at 31·6 lb/in^2 pressure, is used. Ariel I, first flown in 1949, and

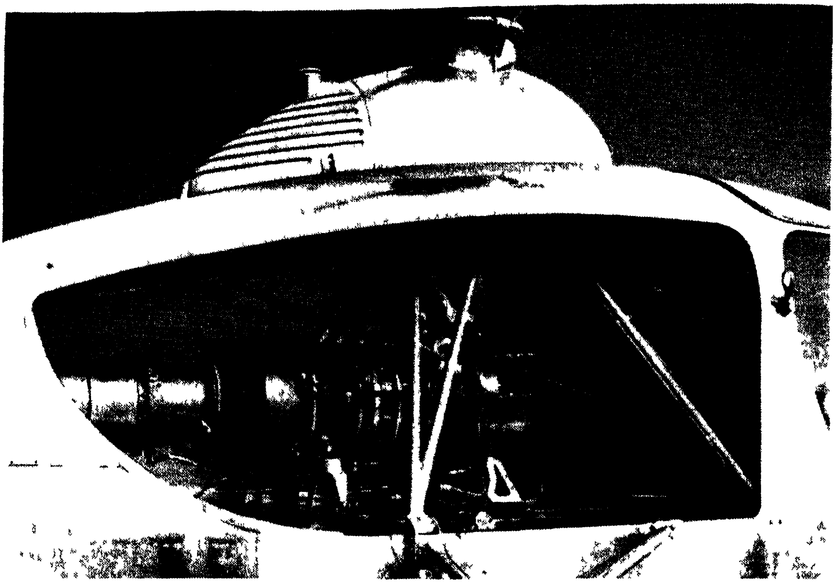


Fig. 15 . 14—Turboméca Arrius I turbine-compressor unit installed in S O 1120 Ariel III

Ariel II had the same combustion jet propulsion system but the air compressor was driven by a piston engine.

Air from the compressor is led through the blades to the small combustion chambers mounted at the tips on the blade centre line. Petrol or kerosine fuel is delivered under low pressure to the rotor hub and injected, under considerable pressure due to centrifugal force, directly into the combustion chambers. The mixture is ignited by an electrical system—high-tension magneto, distributor and spark plugs—which functions continuously to ensure uninterrupted combustion. This method of propulsion is claimed to be preferable to ramjets or pulsejets as it gives easier starting and a more economical specific fuel consumption rate Furthermore, as it requires only small wing-tip chambers, resistance to autorotation is materially lessened and there is no risk of losing rotation in a forced descent in the event of turbine or compressor failure or interruption of fuel supply. The rotor diameter is 35 ft 5 in.

Efflux from the turbine is conducted to the tail of the fuselage to give a small supplementary thrust for forward propulsion. Tailgate vanes interconnected with the rudder are arranged to deflect the gases through lateral orifices to assist steering control.

Empty and gross weights are 1,500 lb and 2,750 lb. Cruising speed is 105 m.p.h. and the service ceiling 14,400 ft. Carrying a disposable load of 660 lb, the range is 155 miles.

The helicopter designed by Mr. L. G. Frise and now under active development by Percival Aircraft uses a Napier-built low-pressure, low-temperature, gas jet drive. A good specific fuel consumption is obtained by compression at relatively high ratio and the whole of the products of combustion pass through the rotor head and blades to the tip jets.

A twin-engined version, P.74, is to be constructed first. Eight passengers will be carried, four in forward-facing seats and, in the aft section, four in rearward-facing seats. Cruising speed is expected to be 120 m.p.h. with a gross weight of 7,750 lb.

In this test vehicle the rotor diameter is 56 ft 6 in. The tip jets are automatically adjustable to allow for single-engine cruising and also for afterburning, which adds considerably to the load that can be lifted when required under special conditions. The amount of heat which is allowed to pass into the skin of the blades is sufficient only to prevent icing. Means are being developed to reduce blade drag by boundary layer control, effected by using the column of gas within the blade. This will also be used to assist in increasing the forward speed of the helicopter by controlling the stall of the blades.

The main problems of ducting of the hot gas into and along the blades with as little loss as possible whilst sealing against rotational and cyclic movement have now been solved. Tests will be run at first from a ground gas producer to determine the overall efficiency of the cycle.

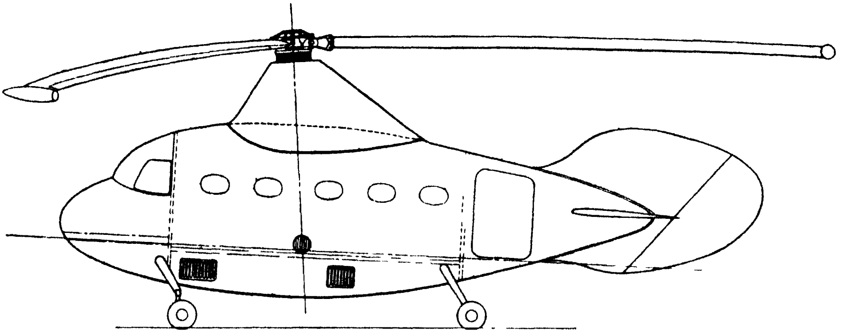


Fig. 15 : 15—Percival helicopter with low-pressure, tip-jet drive.

It is claimed that this cycle gives a very much quieter drive than any of the alternative tip-drive systems or of shaft-driven, piston-engined types.

To describe the compromise type embodying features of the helicopter, autogiro, and fixed-wing aircraft the somewhat clumsy designation of 'convertiplane' has been coined. Such craft have small fixed wings, conventional tail surfaces and one or more power units driving the rotor and a forward-facing airscrew. They are receiving marked attention in several countries and experimental models show great promise, offering the advantages of helicopter performance with substantially improved forward speed. As with more orthodox helicopters, rotor propulsion by blade-tip jets obviates the need for torque reaction control, and turbine power units save weight and reduce vibration.

The original Fairey Gyrodyne had a single anti-torque airscrew mounted on the starboard wing. In forward flight the airscrew provided thrust and the fixed wing furnished sufficient lift to permit the rotor to be unloaded and thereafter autorotate. A second Fairey Gyrodyne has undergone conversion to jet propulsion of the rotor. The piston engine drives a compressor feeding air to blade-tip jets. This machine is being used experimentally for development flying in connection with a more ambitious Fairey project for a 40-passenger convertiplane, the Rotodyne.

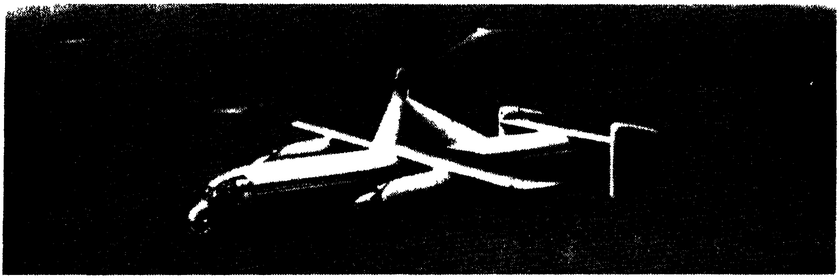


Fig. 15 : 16—An artist's impression of the Fairey Rotodyne.

Although models of the Rotodyne have been exhibited the final configuration has not been disclosed. To power such a craft several alternative installations are possible. Piston engines or turbines could drive compressors for the rotor jets and provide shaft power for airscrews on the wings ; a fuselage-mounted turbo-compressor system could supply air to the rotor and to the combustion chambers of wing-mounted turbines driving airscrews ; or the compressors of modified turboprop units on the wings could be tapped to feed air to the rotor. In any case single-engine safety will be ensured. When hovering at least 90 per cent of the power developed will be applied to the rotor and the airscrews take only sufficient to afford directional control.

Built to the order of the Ministry of Supply, the first Rotodyne will be powered by two wing-mounted Napier Eland turboprop units. Provisional details are for a four-bladed rotor of 90 ft diameter and a fixed wing of 46 ft 6 in span. Forty passengers and their baggage will be carried at a cruising speed of not less than 150 m.p.h. and the still-air range will be about 250 nautical miles.

The first turbine-propelled convertiplane to reach the flying stage of

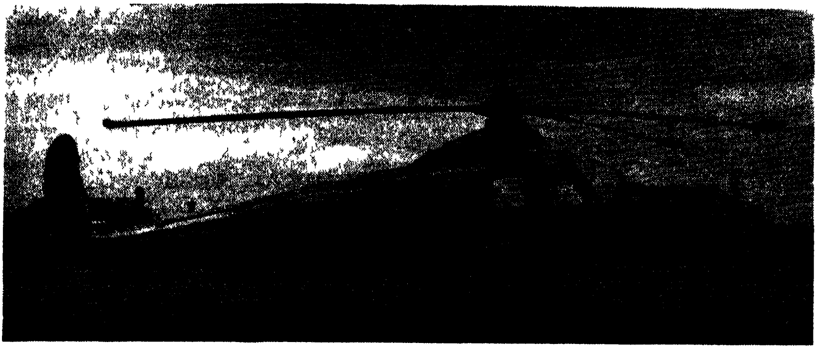


Fig 15 17—S.O. 1310 Farfardet convertiplane.

development was the French S O. 1310 Farfardet in early 1953. In a number of features it resembles the Ariel III jet helicopter built by the same organization but incorporates stub wings and a second turbine unit mounted in the nose of the fuselage and driving a tractor airscew. For rotor propulsion by air jets a Turboméca Arrius II turbine-compressor set is installed in the fuselage behind the cabin. The Arrius II is a 394 b h p. unit as compared with the 276 b.h p. Arrius I set of the Ariel III helicopter. Arrangements for the turbine efflux to be used to assist directional control are similar to those described for Ariel III. Rotor diameter is 36 5 ft and the fixed wing has a span of 19 ft.

Airscrew drive is by a Turboméca Artouste II 394 b h p. single-shaft turbine. With pilot and two passengers a forward speed of 150 m p h. and a range of 250 miles is claimed

Another convertiplane, the McDonnell XV-1, is a high-wing monoplane of 26 ft span, with a twin-boom tail unit. A 550 b h p. Continental

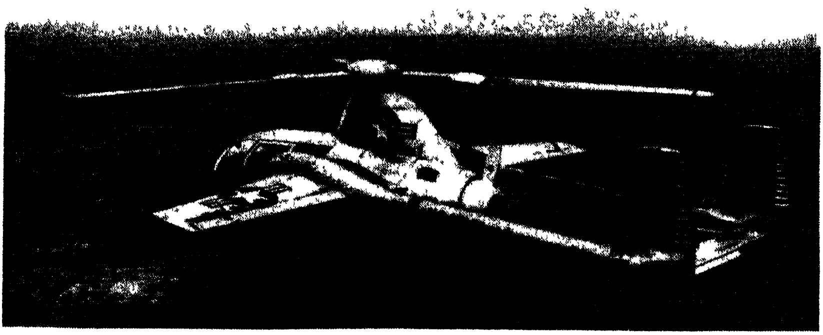


Fig. 15 : 18—The McDonnell XV-1 has mechanical drive for the pusher airscrew. Rotor operation is by blade-tip combustion gas jets.

radial engine mounted in the tail of the fuselage drives a pusher airscrew for forward propulsion and also a compressor supplying air to fuel burners at the tips of the three-bladed rotor. A clutching arrangement enables the airscrew to be disengaged, braked and feathered during rotor operation, while the compressor can be declutched and all power diverted to the airscrew for high forward speed.

CHAPTER 16

Turbines for Road Vehicles

IN view of the long interest in the gas turbine for stationary and industrial service and the amazingly rapid advance of aircraft units it may appear somewhat surprising that relatively little progress has been made in applying the turbine to road vehicles. During the Second World War the turbojet offered substantial and immediate advantage for military aircraft and was therefore intensively developed. No comparable advantage could be expected on land vehicles since the highly developed piston engine meets virtually all requirements with satisfactory reliability. Only the Germans, with their characteristic flair for new weapons and equipment, seriously investigated the potentialities of gas turbines for heavy tanks and special vehicles needing compact high-output power units.

The possibility of obtaining improved power : weight and power : volume ratios in a unit using a relatively low quality fuel was certainly attractive. Subsidiary advantages were the prospects of cheaper and quicker production, simpler installation, easier maintenance and a less complicated transmission system. Compared with an air-cooled aircraft engine, as used in some military vehicles, the supply of clean air presented new problems. A 1,000 h.p. piston engine may consume air at a rate of about 2·2 lb/sec for fuel combustion and require about 11·0 lb/sec for cooling. Only the combustion air needs to be filtered but this must be done thoroughly if a reasonable working life is demanded. The air intake of a turbine unit of similar output would be, say, 22·0 lb/sec, all of which must be cleaned. Unlike the piston engine, the turbine has no rubbing surfaces but the high velocity of the air stream through the unit makes essential the removal of dust.

The absence of any external cooling system greatly simplifies installation, reduces vulnerability, lessens the warming-up period and effects a considerable economy of space. This last advantage may well be completely offset if, in order to improve thermal efficiency, a heat exchanger is installed to recover some of the heat normally lost to atmosphere in a simple, open-cycle unit. Tubular heat exchangers are necessarily of considerable bulk if heat recuperation is to be effected on a worth-while scale. Without a heat exchanger a low-power traction turbine, at that stage of development, could hardly be expected to achieve an overall thermal efficiency of 20 per cent.

According to Lt.-Col. R. H. Bright, the German Government was directly interested in the traction-type gas turbine only for specific military purposes ; armoured vehicles, locomotive and marine applications with power outputs ranging from about 1,000 h.p. to 600 h.p.* A small unit of 320 h.p. intended for normal-type commercial vehicles was not officially sponsored and represented a private venture. All these projects would appear to stem from a detailed design study of a straightforward, open-cycle unit of 1,000 h.p.

Features of this design (Fig. 16 : 1) are a nine-stage axial compressor, a diffuser duct, an annular combustion chamber of the type used in the BMW 003 aircraft unit, a three-stage turbine and a diffuser exhaust. The compressor rotor is built up of light-alloy discs clamped by through-bolts and both rotor and stator blades are die-forged in light alloy. Designed with 50 per cent reaction, the compressor has a pressure ratio of 4·5 : 1 with the axial air speed falling from about 330 ft/sec at entry to 130 ft/sec in the diffuser duct.

A multi-disc construction is also used for the turbine and both rotor and stator blades are hollow and internally air cooled. Gas temperature is approximately 800° C and from 7 to 8 per cent of the air mass flow is utilized for blade cooling. The turbine is held up to the compressor by means of an axial draw-bolt and the complete rotating assembly is mounted on three ball bearings, the centre one being of the self-aligning type. Designed maximum speed is 14,000 r.p.m. and a straight reduction gear from the entry end of the compressor reduces this speed to 3,000 r.p.m. at the coupling. The efficiency of the turbine is estimated to be about 80 per cent, including losses arising from the introduction of cooling air through the blades.

It appears that the Germans investigated five different schemes, illustrated in Fig. 16 : 2, for this traction unit. The first is the simplest arrangement but was regarded unfavourably owing to the torque characteristic being very unsuitable for traction duties and inferior to those of a reciprocating engine. In a simple compressor-turbine combination the available torque falls rapidly with decrease of speed from the design maximum and, as a consequence, a relatively elaborate and complicated transmission system would be required.

In the second scheme, separate turbines are used to drive the compressor and to furnish useful power output. This arrangement obviates the poor torque characteristics of the simpler unit and, in fact, possesses a torque characteristic superior to that of the piston engine. The compressor is the same as in the first scheme but 35 per cent of the air delivered is tapped off to feed a separate combustion chamber for the power turbine (13·0 lb/sec for compressor turbine, 7·6 lb/sec for power turbine). The remaining air passes through a combustion chamber to the

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* Paper presented before the Institution of Mechanical Engineers.

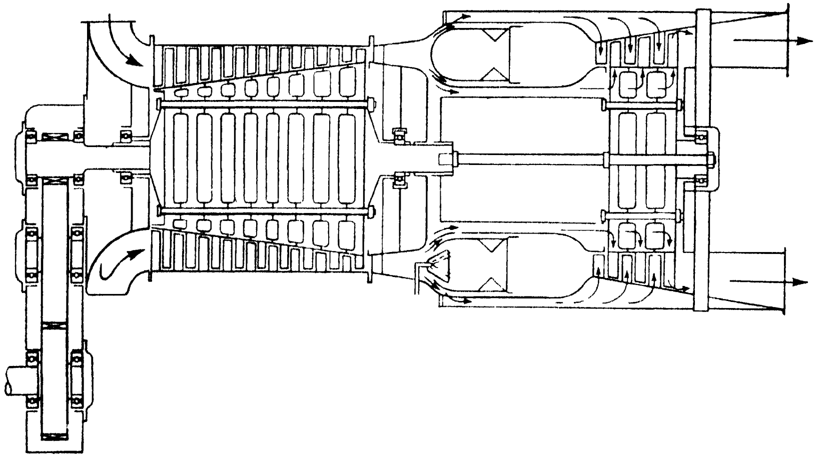
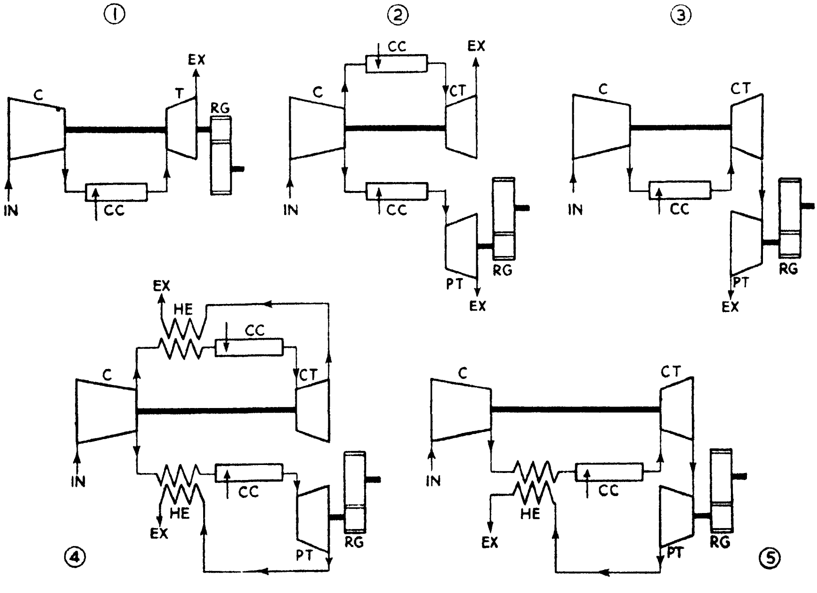


Fig. 16 · 1 - Basic design for German traction turbine.



C. Compressor
CC. Combustion chamber.
HE Heat exchanger.
RG. Reduction gear.
T. Turbine.
CT. Compressor turbine
PT. Power turbine.

Fig. 16 : 2—Five arrangements suggested for German traction units.

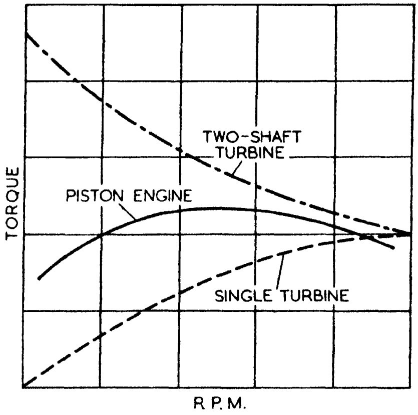


Fig. 16 : 3—Relative torque curves of normal piston engine, single turbine unit and two-shaft turbine plant.

compressor turbine which is of lower output than in the first scheme as it serves solely to drive the compressor. Driven at 14,000 r.p.m. the compressor is entirely independent of the power turbine designed to run at 20,000 r.p.m. A torque ratio of at least 2 : 1, stall to maximum r.p.m., may be expected from an independent power turbine and consequently a simple two-speed gear will furnish sufficient variation of torque for the vehicle. It is claimed that the turbine may be held in the stalled condition without suffering ill-effects, providing the supply of cooling air through the hollow turbine blades is maintained.

The third scheme is similar to the second but only a single combustion chamber is used and the two turbines are arranged in series. This may offer some advantage as regards first cost and possibly space occupied but, on the other hand, demands the location of the power turbine in close proximity to the compressor turbine if large-diameter insulated ducting is to be avoided. It can offer slightly higher overall efficiency at low loading or when idling as the compressor cannot be run at constant speed as in the second scheme.

Fourth and fifth schemes are merely the second and third arrangements with the addition of heat exchangers to recover some of the heat normally exhausted from the turbine to air. By these means it was hoped to reduce the specific rate of fuel consumption to about 0·33 lb/b.h.p./hr and 0·5 lb/b.h.p./hr for 1,000 h.p. and 800 h.p. respectively.

For the 320 h.p. unit a two-shaft design was adopted and it was expected that the plant could be built with a maximum diameter of about 27·5 in. A six-stage axial-flow compressor would handle an air flow at the rate of about 14 lb/sec with a pressure ratio of 3·7 : 1. The two-stage primary turbine driving the compressor would run to 21,000 r.p.m. with a temperature drop from 750° C to 565° C, whilst the single stage power turbine, running at a maximum of 7,500 r.p.m., would exhaust at 540° C. The specific fuel consumption was estimated at 1·7 lb/h.p./hr. If a heat exchanger was incorporated, the overall diameter would be raised to about 39·5 in, but the rate of fuel consumption could be cut to about 0·9 lb/h.p./hr.

It must be noted that none of these projects was ever completed nor were the major components, except for such items as the BMW combustion chamber, ever built. The deteriorating position in Germany toward the end of the war, and a variety of factors other than the technical, prevented the production of experimental units.

With the cessation of hostilities attention could be devoted to projects of potential commercial value. A good case could be made for turbine propulsion of goods and passenger vehicles and, with less certainty of high-grade private cars. One British motor-car manufacturer, the Rover Co. Ltd. of Solihull, Birmingham, has designed and built several experimental 100 h.p. units. There is some significance in this enterprise. Not only is the Rover Co. one of the oldest-established car manufacturing concerns but it also possesses extensive experience in the design of aircraft gas turbines. During the war it was entrusted with the development of the early Whittle jet propulsion units and, in conjunction with Joseph Lucas, Ltd., introduced the ' straight-through ' combustion system which replaced the earlier ' reverse-flow ' system and became a feature of many successful power units. Later, when the resources of the Rover Co. were required for the production of tank engines, the aircraft turbine development was continued by the Rolls-Royce Co.

The interest of the Rover engineers, however, was maintained and when conditions permitted they investigated the application of turbine power units to road vehicles. An assessment of the possibilities proved attractive and a decision was made to build a turbine that could be installed in the currently-produced Rover chassis in place of the six cylinder, 2,100 cm^3 reciprocating engine. The first example, shown in Fig. 16 : 4, was specifically a ' car ' turbine. Compactness was attained by the use

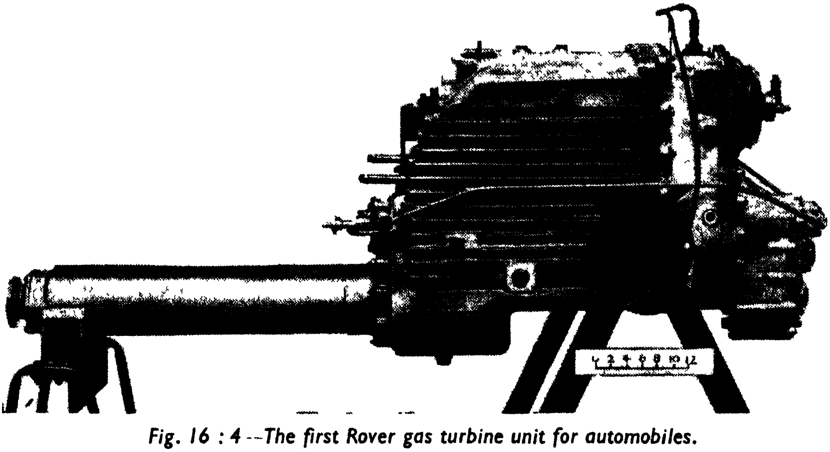


Fig. 16 : 4—The first Rover gas turbine unit for automobiles.

of a centrifugal compressor and the drive shaft extension projecting to the rear was provided solely to enable the unit to be coupled to the standard propeller shaft in the Rover chassis. Approximate overall dimensions were : length 34 in, height 20 in and width 18 in. Complete with its built-in 20 : 1 reduction gear the weight was 475 lb, effecting a saving of about 300 lb when compared with the 75 b.h.p. piston engine with clutch and gearbox.

The Rover power unit was of considerable technical interest, being of the two-shaft type and arranged for the eventual incorporation of a heat exchanger in the gas cycle. Each turbine rotor had a disc with integral blades machined from a solid forging of Nimonic alloy An early rotor, with a disc less than 5 in diameter, had a designed maximum speed of 55,000 r.p.m. but later types had lower operating speeds. Maximum output on the test bed was approximately 100 b.h.p. The turbine was started by a car-type electric starter driving through a 5 : 1 increasing gear and by means of an electric igniter plug the unit was 'lit-up' at about 10,000 r.p.m. As on an aircraft unit, actuation of a single switch commenced a timed sequence of starting operations, including a short period during which the starter sustained the turbine until a speed in excess of 15,000 r.p.m. was reached, and then automatically cut out.

History was made on March 8, 1950, when the Rover turbocar, the first gas turbine-driven automobile in the world, was submitted to an officially observed R.A.C. test at the proving ground of the Motor Industry Research Association. The object of the test was not to attain maximum speed or to establish a record, but merely to demonstrate the stage of progress achieved by the Rover Company in the application of the gas turbine as a power unit for road vehicles. The following notes are quoted from the R.A.C. report :

> 'The power plant which is mounted immediately ahead of the rear axle consists of a centrifugal compressor, with dual combustion chambers, having a single-stage compressor turbine, and an independent power turbine. The latter is positively coupled through gearing, incorporating a reverse gear, and drives a conventional rear axle. The fuel used was kerosine.
>
> 'The time taken to start the power unit and to run up to idling speed was 13-1/5 seconds, and the car moved forward in a further 3-2/5 seconds. Control is simple from standstill to maximum speed, and is solely by means of the accelerator pedal, the only other driving control being the brake pedal.
>
> 'No attempt was made to attain maximum speed, but during the course of the Test a speed exceeding 85 m.p.h. was readily attained, at which speed the compressor-turbine revolution counter indicated

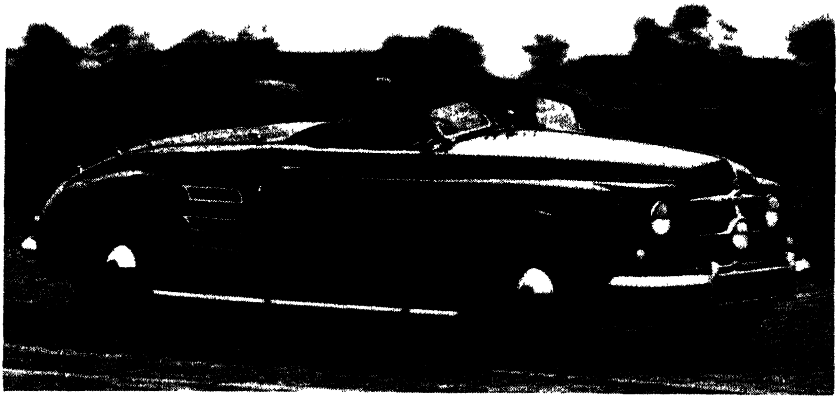


Fig. 16 : 5 – The first turbocar in the world. The Rover at speed on the Silverstone track, March, 1950.

> 35,000 r.p.m. In a test of acceleration from a standstill, the car smoothly attained 60 m.p.h. in 14 seconds.
>
> ' Although no provision for silencing the exhaust was observable, the volume of noise was not excessive or unpleasant, but was naturally accentuated during acceleration.'

The car is shown in Fig. 16 : 5 travelling at speed during a demonstration at the Silverstone track on the day following the test.

Development was actively continued and after an interval of slightly more than two years a Rover turbocar was subjected to high-speed tests under international regulations on the Jabbeke road in Belgium. In this instance the object was to establish the first internationally recognized world speed records for the type. The car was the one used for the R.A.C. test but was modified by the substitution of Girling disc brakes for the standard drum brakes to facilitate stopping after high-speed runs. It had the standard chassis, steering and suspension of the Rover 75 production cars. While of the same basic type, but with one instead of two combustion chambers, the turbine installed, Fig. 16 : 6, was a more highly developed unit which, with a compressor speed of 39,500 r.p.m., had developed 230 b.h.p. on the dynamometer.

For the speed tests on June 25, 1952, the turbine was fitted with a 4·875 : 1 reduction gear which in conjunction with a rear axle ratio of 3·64 : 1 gave an overall ratio of 17·745 : 1. With these gears and with the compressor speed held to 37,500 r.p.m. the highest speed recorded—the mean of two runs in opposite directions—was :

Flying kilometre	140·422 m.p.h.
Flying mile	137·403 m.p.h.

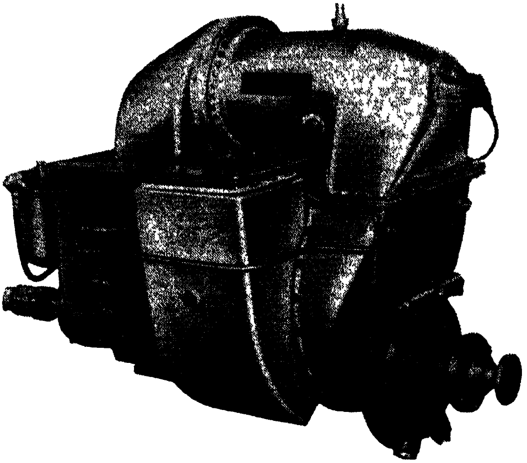


Fig 16 6 The Rover T 8 power unit has a maximum output of 230 b h p

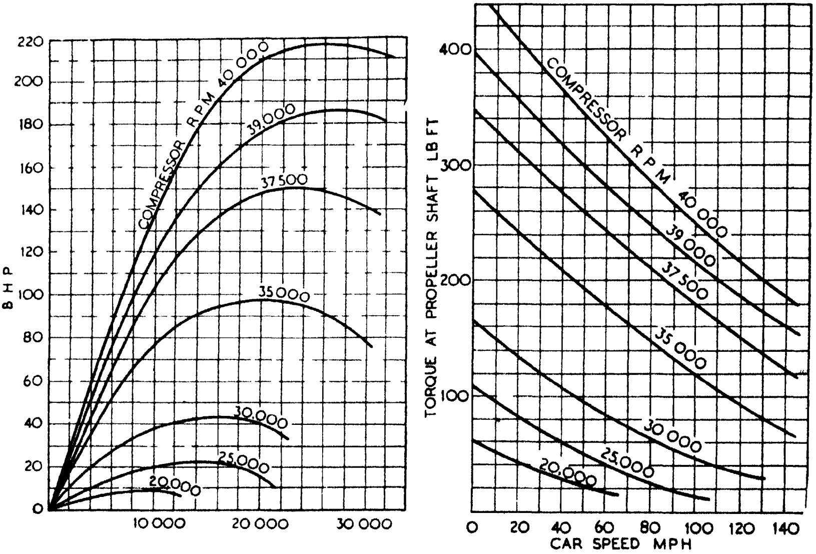
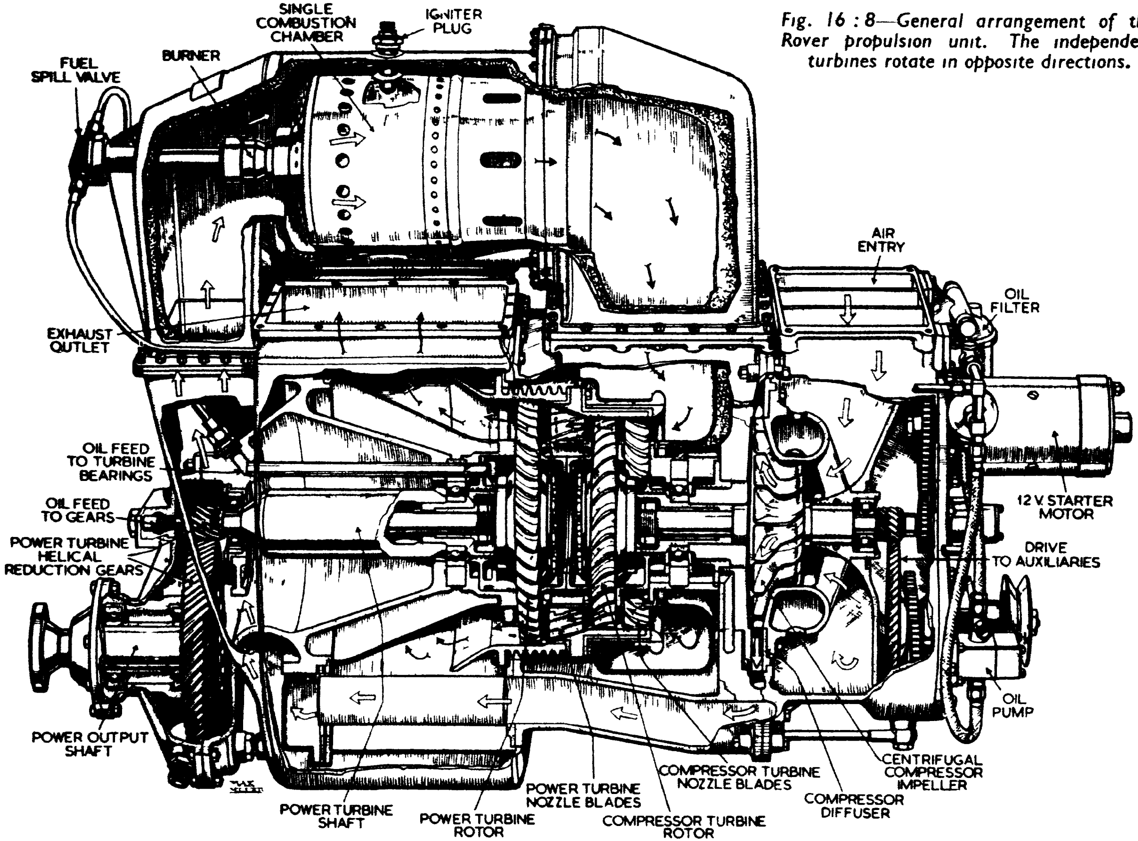


Fig 16 7 —Power output and torque curves of the Rover T 8 unit.

Fig. 16 : 8—*General arrangement of the Rover propulsion unit. The independent turbines rotate in opposite directions.*



As there was ample power in hand the rear axle ratio was changed to 3·275 : 1 giving a new overall ratio of 15 965 : 1. The following day further runs were made to establish mean speeds of :

Flying kilometre	151 965 m.p.h.
Flying mile	151 196 m.p.h.

From a single vertical intake duct the compressor at the rear of the unit delivers air to the passage at the front leading to the single combustion chamber mounted overhead. Of forged light alloy, the compressor impeller is 8 75 in diameter and is furnished with curved entry vanes of nickel steel. In this instance the flow is by way of a series of plain 1 75 in diameter tubes used in substitution for the heat exchanger which is under development.

Fuel is sprayed into the combustion chamber by Lucas equipment operating at approximately 300 lb/in^2 pressure. Regulation is by a spill system controlled from the accelerator pedal. The Nimonic 75 flame tube is housed in an aluminium-sprayed, mild-steel casing. Both compressor turbine and power turbine discs of forged Nimonic 80 are 6·3 in diameter but the integral blading gives overall diameters of 7·88 in and 9·3 in respectively. The compressor turbine rotor has 35 blades and operates with a gas entry temperature of about 900° C.

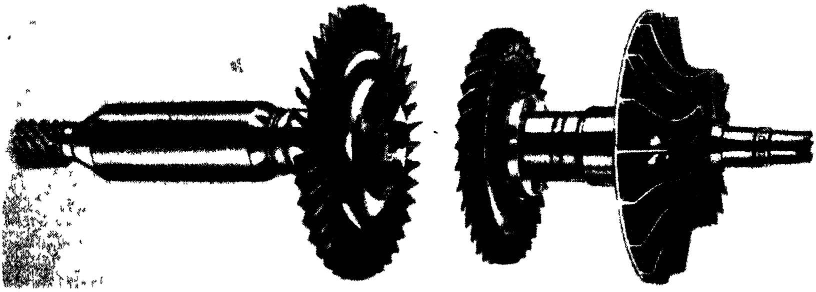


Fig. 16 : 9—Independent co-axial rotating assemblies of Rover car turbine.

At the front end the power turbine drives the output shaft through a single helical reduction gear. By means of a 1 : 1 transfer box the drive is reversed and a short propeller shaft runs below the unit to the rear axle. Auxiliaries are mounted on the rear cover and driven through helical and spur gears from the end of the compressor shaft. The unit weighs about 500 lb and the complete vehicle scales 25 cwt.

Starting is effected by a single switch which energizes the Lucas 12 V starter motor, a small electric pump to raise fuel pressure, and the surface discharge igniter plug. Light-up occurs at about 3,000 r.p.m. of the compressor, at 10,000 r.p.m. the compressor delivery pressure automatically

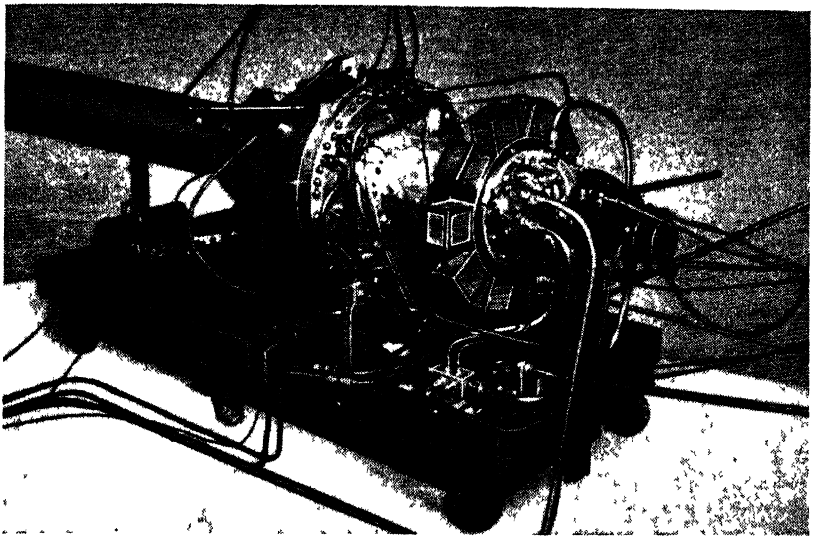


Fig 16 10 Centrax compressor and turbine unit on test.

actuates a switch to cut out the three auxiliaries and to withdraw starter dogs, and the unit runs up to the controlled idling speed of 13,000 r p m The slight torque at this speed may, under some conditions, be sufficient to move the car which is normally held on the brake. Reference to the curves (Fig 16 : 7) shows that a very high torque is available for starting and low speeds, the torque ratio being about 2 5 : 1

Another unit designed specially for road vehicles, the Centrax, was exhibited at the British Industries Fair, 1948. Sponsored by Messrs Barr, White and Leach, former members of the Power Jets organization, it has a design output of 160 h p. This unit, the compressor-turbine element of which is shown on test in Fig. 16 : 10, is also of the two-shaft

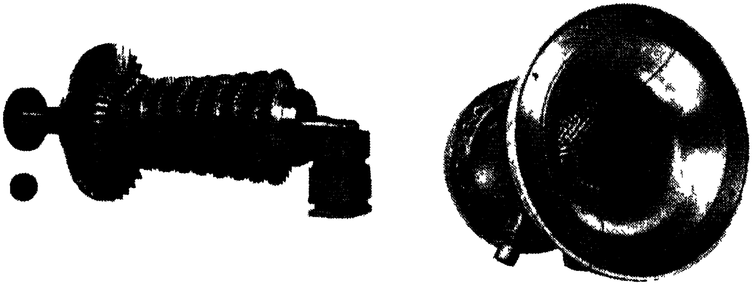


Fig. 16 : 11—Rotor and stator casing of Centrax compressor

type, but having the relatively high pressure ratio of 5·9 : 1 it is not fitted with a heat exchanger. To secure this ratio the compressor has eight axial-flow stages and a final centrifugal stage. Air enters the compressor through ten box-type gauze filters; the throughput being 2·15 lb/sec. The rotor and its stator casing are shown in Fig. 16 : 11, scaled by a penny, 1-1/16 in diameter, placed alongside.

A two-stage turbine drives the compressor at a speed of 43,000 r.p.m. and the single-stage power turbine runs at a maximum of 35,000 r.p.m. Both turbine rotors are 6 in diameter. Fuel is burned in an annular combustion chamber having seven flame tubes and maximum gas temperature at the turbine entry is 827° C. Kerosine or diesel oil is used as the fuel and the estimated rate of consumption is 0·7 lb/b.h.p./hr. Overall dimensions are 17 in diameter and 60 in length and the dry weight 250 lb.

A turbine of a size suitable for a road vehicle is likely to be more difficult to construct and to be less efficient in operation than a high-powered unit for aircraft. The reasons are both mechanical and aerodynamic. It is possible to scale down the dimensions of the components of a 2,500 h.p. aircraft turbine but not practical to scale down the limits of tolerance on size, shape, weight and surface finish. Nor can the working clearances be correspondingly reduced. For units of similar type, power output is approximately proportional to the square of a linear dimension and consequently the dimensions of a 150 h.p. car turbine will be about one-fourth those of a 2,500 h.p. aircraft unit.

As a result of the small dimensions the Reynolds' number conditions for compressors and turbines will be lowered and gas friction losses will be increased. However, as operating velocity will be of similar order, density the same, and viscosity the same, size is the only factor exercising an important influence. At high altitude the larger aircraft unit may work at a lower effective Reynolds' number than the road vehicle turbine due to the reduced density. On the other hand, at relatively low Reynolds' number there is the possibility of laminar separation of flow from the blade profile and some advantage might be gained by inducing a transition to turbulent flow upstream of the separation point. A relative roughening of the blade surface might achieve this effect and thus a necessarily inferior surface finish may be turned to good account. In this connection it may be mentioned that Dr. A. A. Griffith has patented, British No. 580,806, a blade on which the surface is either treated to produce a relative roughness or is left in the 'as cast' condition to promote turbulence in the boundary layer and delay separation. In these small compressors and turbines it would seem reasonable to expect individual efficiencies about 5 per cent lower than those attainable on large components and an overall efficiency for the unit lowered by, perhaps, 16 to 20 per cent.

A small unit must operate at about the same pressure ratio as a large one to obtain comparable efficiency and it therefore becomes necessary to maintain the same peripheral velocity. Rotational speed varies inversely with the linear dimension or, in other words, with the square root of the power output. A 150 h.p. unit will be run at 40,000 r.p m. or more and will necessitate the use of high quality anti-friction bearings. Experience with the bearings of superchargers for aircraft engines suggests that no trouble need be expected providing the dimensions are kept down and the peripheral velocity of the balls or rollers is not increased. In this respect the road vehicle turbine, operating for most of its working life at partial loads lower than the normal cruising load of the aircraft unit, has some advantage. Only on comparatively rare occasions will the maximum speed of rotation be required.

Gearing to reduce the speed of the power turbine at a ratio of 10 : 1 to 15 : 1 will call for meticulous design and workmanship, but again the aircraft engine supercharger gearing can furnish useful data. It will be necessary to arrange for a two-stage reduction and probably helical gears will be desirable to ensure quiet operation.

A number of manufacturing difficulties arise owing to the small dimensions and the techniques evolved for the large aircraft unit cannot always be applied. Turbine blading is a typical example. The blades must be of the same order of accuracy in profile, section and twist to obtain comparable efficiency and are thus liable to be costly. It is impracticable to use the familiar fir-tree root fixing in such small sizes and other methods must be sought. Integral blades, machined directly in the periphery of the disc, have been proved practical by the Rover Co. Given adequate production equipment, this method would appear to offer a solution to a difficult problem.

The fuel system is directly affected as a much smaller quantity has to be atomized, particularly if a plurality of chambers is employed. A remotely-sited single chamber is of dubious advantage as it necessitates bulky ducting and may give rise to indifferent temperature distribution at the turbine entry. To secure good atomization when idling or running at low loading a small fuel orifice and high pressure will be required. The alternative may be to use a pump of more modest pressure and to control the supply by spilling the delivery.

On a road vehicle turbine the compressor must be equipped with an air filter. This applies with particular emphasis if an axial compressor is used as, apart from the danger of erosion, dirt tending to accumulate on the blading may seriously impair the efficiency. The filter must necessarily have a large flow area to avoid an uneconomical pressure loss at the compressor entry, but can serve a useful purpose in reducing noise.

A problem associated with the road vehicle turbine is the difficulty of starting in the event of a failure of the normal electric motor or battery. A start cannot be made by running the vehicle down a hill as there is no

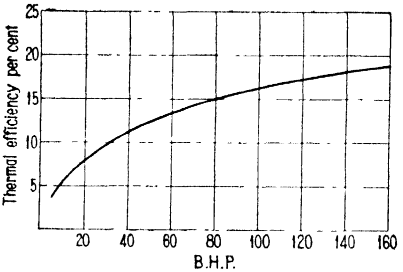


Fig. 16 : 12—Thermal efficiency v. b.h.p.

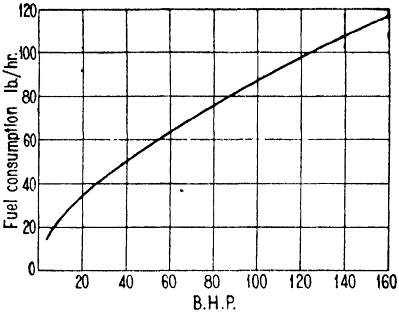


Fig. 16 : 13—Fuel consumption v. b.h.p.

connection between the road wheels and the compressor turbine. It would not appear practical to crank through an increasing gear to reach a lighting-up speed. Possibly an inertia starting device could be incorporated or an emergency clutch or coupling to link the compressor turbine to the power turbine and thus permit driving from the road wheels, might suffice.

As an example of the application of the gas turbine to a heavy vehicle, Mr. R. R. H. Barr made an examination of the Centrax two-shaft unit without heat exchanger for a double-decked city omnibus. The curves, Figs. 16 : 12 and 16 : 13, for thermal efficiency and fuel consumption plotted against power output are based on the following full-power figures :

Output	160 b.h.p.
Air mass flow	2·15 lb/sec
Pressure ratio	5·93 : 1
Turbine inlet temperature	827° C
Overall compressor efficiency	77 per cent
Compressor turbine efficiency	85 per cent
Power turbine efficiency	85 per cent
Combustion efficiency	98 per cent
Intake pressure loss	0·1 lb/in²
Combustion pressure loss	3·0 lb/in²
Exhaust ducting loss	60 per cent of turbine exit dynamic head

Assuming the characteristics of a typical double-decker with 60 passengers are :

Weight, fully laden	20,000 lb
Frontal area	100 ft²
Aerodynamic drag coefficient	0·002
Rolling resistance	30 lb/ton

the drag of the laden bus will be :

$$\frac{30 \times 20{,}000}{2240} + (0{\cdot}002 \times 100 \times V^2)$$

or

$$268 + 0{\cdot}2V^2$$

and the drag h.p.:

$$\frac{268 \times V \times 88}{550 \times 60} + \frac{0{\cdot}2 \times V^3 \times 88}{550 \times 60}$$

or

$$0{\cdot}714V + 0{\cdot}000533V^3$$

Drag h.p. plotted against vehicle speed is given in Fig. 16 : 14 and if this curve is combined with that of Fig. 16 : 13 a performance curve in which fuel consumption in m.p.g. plotted against vehicle speed is obtained, Fig. 16 : 15. This is for steady running on the level and consumption will be somewhat higher if frequent stops and restarts are taken into account.

A 100 b.h.p. diesel-engined bus averages between 8 and 10 m.p.g. and a 100 b.h.p. petrol bus about 5 m.p.g. It would be possible to reduce the fuel consumption of the turbine to about 7 m.p.g. if a heat exchanger of 60 per cent thermal ratio was employed.

One property of the two-shaft turbine gives it a great advantage over the piston engine. An efficient torque conversion mechanism which obviates the need for a change-speed gearbox and clutch is an inherent feature of the design. Consider the case when the power turbine is supplied with constant inlet gas conditions from the compressor-turbine unit which is held at constant speed by maintaining a constant fuel supply. Fig. 16 : 16 shows that as the load is increased the speed of the turbine falls. The load can be increased until the power turbine ceases to revolve and in this stalled condition the torque may be three times the value at maximum r.p.m.

The efficiency of this mechanism can be judged by reference to Fig. 16 : 17 showing turbine efficiency plotted against speed. Between points A and B, spaced equally about the maximum efficiency point, there is

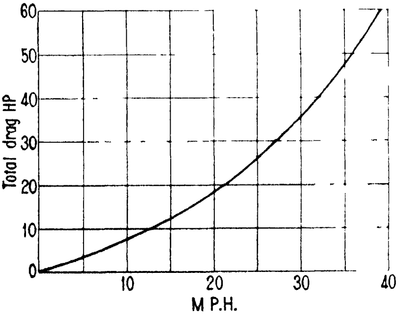


Fig. 16 : 14—Power requirement of 60-passenger bus.

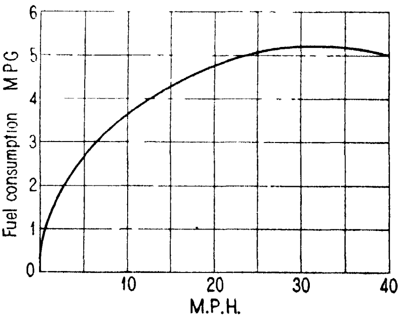


Fig. 16 : 15—Fuel consumption of laden bus.

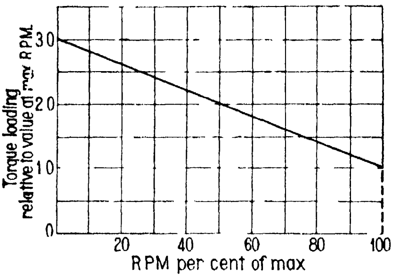


Fig. 16 · 16—Effect of r.p m. on torque, under constant inlet conditions.

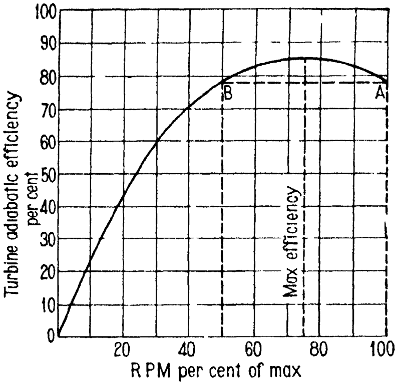


Fig. 16 : 17—Effect of r.p.m. on efficiency, under constant inlet conditions.

2 : 1 speed variation with an efficiency fall-off from the maximum of only about 7 per cent. Thus, if the reduction gearing is arranged so that when the vehicle is running on a level road it is working at point A, then when the load is doubled by the vehicle climbing a hill the turbine accommodates this perfectly smoothly by working at point B.

For all normal running no gear changing is necessary but it may be desirable to fit an emergency low gear for ascending hills of greater gradient than about 1 in 8. Such a gear will need to be used only on rare occasions. Absence of gear changing will lead to an appreciable amount of time saving owing to rapid acceleration between stops, whilst the driver will be relieved of considerable fatiguing effort. Acceleration will be beneficially affected by the reduced weight of the power unit and also by the fact that, even at low vehicle speeds, full torque can be applied in the few seconds required to run the compressor turbine up to full speed.

Control will be simplified as both the clutch pedal and the gear change lever can be dispensed with. Under idling conditions when the vehicle is stationary, the compressor turbine may be revolving at the relatively low speed of, say, 8,000 r.p.m. and not supplying sufficient power to the power turbine to move the dead weight of the vehicle. As the compressor-turbine rotor is speeded up by depressing the accelerator and thus supplying more fuel, the energy supplied to the power turbine rapidly increases and, as soon as it is sufficient to overcome the starting friction of the vehicle, the vehicle moves forward. There is, in effect, a fluid coupling as well as a torque converter.

The principal controls are accelerator and brake. It will be necessary to provide a reversing mechanism and its corresponding control lever, but this is only used occasionally and cannot be regarded as one of the main controls. When operating a turbine-driven vehicle, both hands can be kept on the steering wheel the whole time and one foot can be used for each of the two pedals.

A high thermal efficiency and consequent low specific fuel consumption can be obtained if a high gas temperature at the turbine entry is employed. However, the ability of the turbine blade and disc materials to maintain strength and avoid creep and scaling at high temperature imposes a practical limit in this direction. With present-day alloys a maximum gas temperature at the turbine of approximately 800° C is permissible, and when a working temperature has been selected two courses are open to the designer. The unit may have a relatively high pressure ratio or be arranged with a relatively low pressure ratio but incorporating a heat exchanger. With the first a more elaborate compressor, two-stage centrifugal or axial and final centrifugal, will be required but the alternative type with only a simple compressor will be complicated by the bulky heat exchanger and ducting.

A unit having a pressure ratio of 3 : 1 but equipped with a heat exchanger has a higher efficiency than a high pressure unit with a ratio of 6 : 1. Moreover, it is easier to start and also has a materially improved efficiency at part loads. This is a most important consideration for a road vehicle unit which will run most of its life at part load.

To the U.S.A. goes the credit for operating the first turbine-propelled commercial vehicle. This was the Kenworth articulated 10-tonner, powered by the Boeing 502 turbine, which was publicly demonstrated on April 10, 1950. Developed for marine propulsion and auxiliary power generation from the Boeing 500 turbojet, the unit has a single-stage centrifugal compressor, two combustion chambers, independent compressor and power turbines in series, and a 9·06 : 1 reduction gear. The pressure ratio is unusually low at 3 : 1 and the design, in its current form, does not admit the incorporation of a heat exchanger. As a consequence the specific rate of fuel consumption is too high to be economically justifiable for road transport duty. The aim is to lower the rate by improvement of component efficiencies.

Maximum output of 175 b.h.p. is obtained with a gas temperature

Compressor efficiency ...	... 82·5 per cent
Turbine efficiency	... 85·0 per cent
Turbine inlet temp	... 827°C
Heat exchanger ...	... 60 per cent thermal ratio

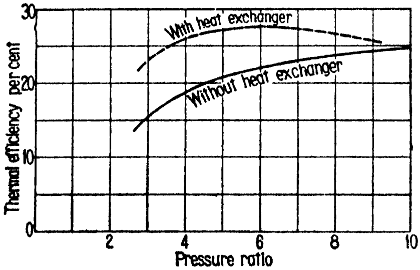


Fig. 16 : 18—Thermal efficiency v. pressure ratio.

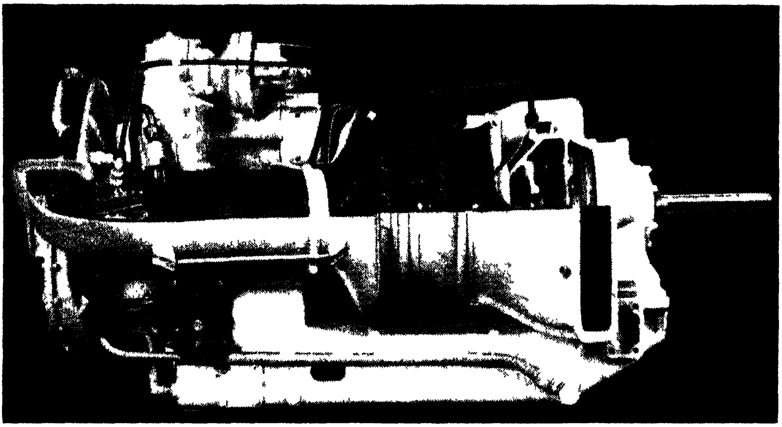


Fig 16 19—Boeing 502 two shaft turbine unit in part section

at the turbine nozzle of 815° C and a compressor speed of 36,000 r p m, giving 22,500 r p m of the power turbine and 2,483 r p m of the tail-shaft Overall dimensions are 1 ft 10 in diameter and 3 ft 6 in long and the weight of the complete unit is 200 lb

On the vehicle the turbine unit saves about 80 per cent of the space occupied and more than 2,500 lb of the weight of the conventional diesel power plant Against these advantages must be set the greater weight

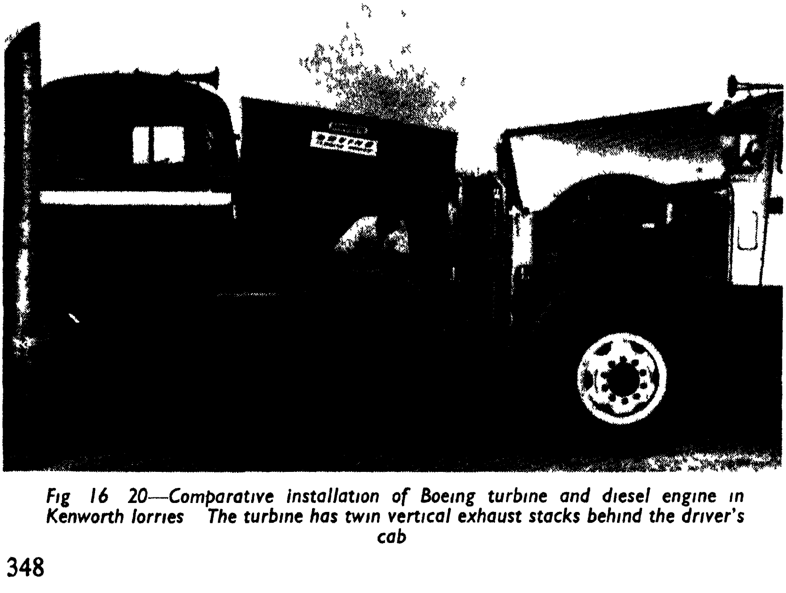


Fig 16 20—Comparative installation of Boeing turbine and diesel engine in Kenworth lorries The turbine has twin vertical exhaust stacks behind the driver's cab

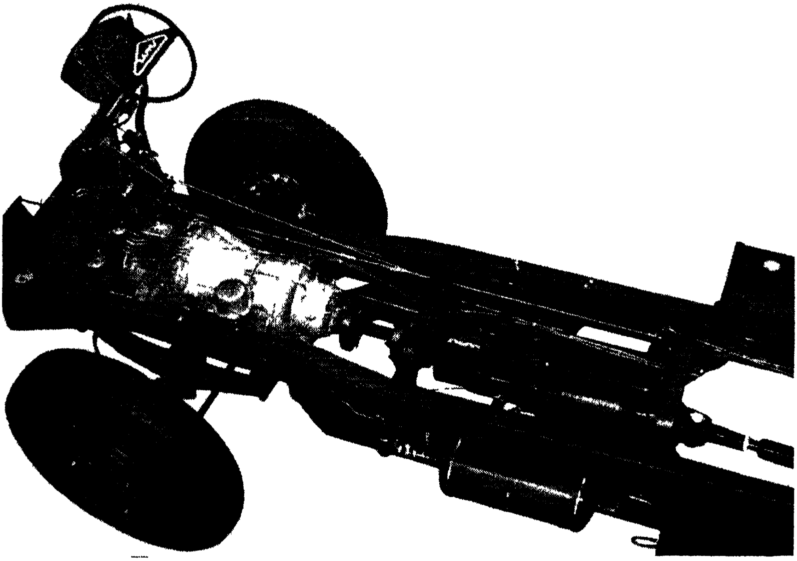


Fig 16 : 21– Turbine power unit and transmission on Laffly 10-tonner chassis.

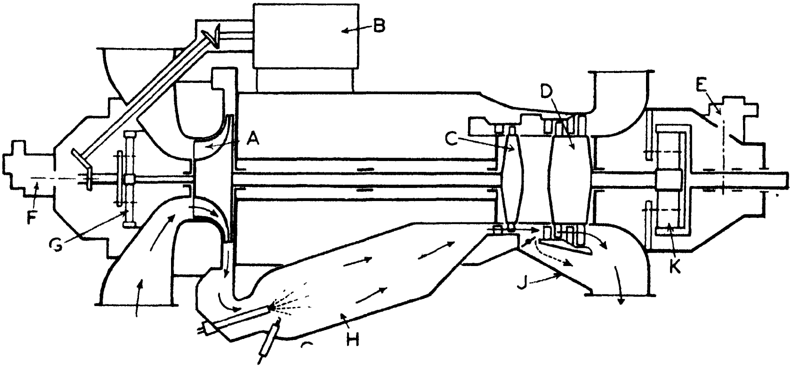
and bulk of fuel to be carried, but a substantial increase of payload should be possible. After 550 hr running, covering 15,000 miles of operation on the road, hauling mostly a gross weight—vehicle and load—of 67,000 lb, it was reported that the average rate of consumption of diesel oil was little more than one mile per gallon.

Another commercial vehicle application, exhibited at the Paris Salon in 1951, may be regarded as a prototype. A Laffly 10-tonner was fitted with a turbine unit rated at 180–200 h.p. The single-stage compressor runs at a maximum speed of 30,000 r.p.m. and the independent, two-stage power turbine up to 24,000 r.p.m. and drives through a reduction gear having a ratio of 8·6 : 1. The centrifugal compressor delivers to two combustion chambers and the fuel for normal operation is diesel oil although petrol is used for starting. A specific rate of fuel consumption of 400 gr/ch/h (0·9 lb/b.h.p./hr) is claimed. The transmission line incorporates a two-speed and reverse gearbox, giving a 3 : 1 reduction for starting and steep gradients, and a seven-disc transmission brake. The brake is operated by the usual pedal controlling the wheel brakes and also automatically on release of the accelerator to simulate the braking effect of the conventional reciprocating engine.

The provision of this transmission brake focuses attention on a disadvantage of the two-shaft turbine unit for road vehicles. Due to the low inertia of the independent power turbine and the virtual absence of friction, the unit cannot be used in the same manner as a reciprocating engine to furnish braking torque to supplement the vehicle brakes. In the absence of an automatic braking device, such as used on the Laffly vehicle, the turbine must be regarded as a free-wheeling unit and when descending a hill, conditions would be comparable to coasting in neutral. Several methods have been proposed to provide braking torque independent of the wheel brakes, including clutching in the main air compressor, utilizing the power turbine as a compressor and driving an independent compressor from the transmission.

At the Paris Salon 1952 a new 100 h.p. turbine was shown in a Gregoire-designed car resembling the currently produced Hotchkiss, but it is intended also, in higher-powered units, for use in passenger and commercial vehicles. Built by SOCEMA (Société de Construction et d'Etudes de Matériels d'Aviation) the unit is of straightforward design and arrangement with a centrifugal compressor feeding three combustion chambers into which petrol is sprayed at a pressure of 70 kg/cm^2 (995·6 lb/in^2). The weight of the complete unit is 264 lb but this, it is claimed, will be reduced.

Maximum operating speeds, limited in each case by overspeed governing devices, are 45,000 r.p.m. for the single-stage compressor turbine and 25,000 r.p.m. for the two-stage power turbine. It will be noted that in the passage between the two turbine rotors a valve is fitted to control a duct allowing the gas stream to by-pass the power turbine. Functional details were not specifically defined but it could be used to facilitate control in either, or probably both, of two ways. To secure rapid



A. Compressor.
B. Starter-generator.
C. Compressor turbine.
D. Power turbine.
E Governor, power turbine.
F. Governor, comp. turbine
G. Auxiliaries reduction gear.
H. Combustion chamber.
J. By-pass conduit.
K. Reduction gear.

Fig. 16 : 22—SOCEMA Cematurbo unit.

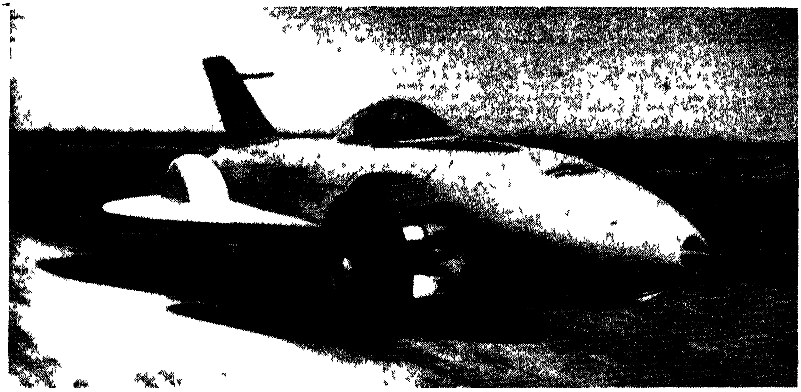


Fig 16 23—General Motors 370 h p. turbine-powered test vehicle.

acceleration of the vehicle it could be opened temporarily to relieve back-pressure on the compressor turbine, thus accelerating the compressor which can then feed more air to burn more fuel without unduly raising the maximum gas temperature. It could also be used when idling the unit, to avoid the necessity of holding the vehicle stationary on the brakes against the torque of the power turbine, or when manœuvring the vehicle. The transmission of this experimental car is relatively elaborate From the incorporated 5 : 1 reduction gear a propeller shaft runs to an assembly located immediately to the rear of the driving seat and comprising a disc clutch, a Cotal electro-magnetic gear unit, a Telma electro-magnetic brake and finally a de Dion-type axle.

As may well be expected a number of the major motor manufacturers are actively investigating the possibilities and prospects of the turbocar. Chrysler, Ford and General Motors in the U.S.A. and Austin in England may be cited in this connection. General Motors early in 1954 exhibited a turbine-powered experimental vehicle but, as will be noted from the illustration, in a form that rendered it hardly practicable for use on public highways. Although the chassis dimensions are not abnormal—wheelbase 8 ft 4 in, track 4 ft 6 in front and 4 ft 2 in rear—the cylindrical body of glass fibre reinforced plastics is 18 ft 7 in long. Developed after studies of a model in a wind tunnel and resembling the fuselage of a fighter aircraft, the body has a sharply pointed nose, large-diameter exhaust outlet at the tail and a single direction-stabilizing fin. Unusual features are small delta-shaped horizontal fins commencing immediately behind the exposed front wheels and extending past the rear wheel enclosures almost to the tail. At the trailing-edges of these horizontal fins are provided aerodynamic brakes, power-actuated from a control on the steering wheel. Extreme width over the fins is 6 ft 8 in.

Access to the vehicle is obtained by way of a hinged upper section carrying the bubble canopy enclosing the driver.

The power unit is of simple open-cycle type, with a charging set comprising a centrifugal compressor driven by a single-stage turbine at 26,000 r.p.m. and delivering air at a pressure of 3·5 atm to two combustion chambers. Kerosine is used as fuel and the gas temperature at the turbine nozzle is 815° C. No heat exchanger is provided. The independent power turbine runs at about 13,000 r.p.m. and drives the rear wheels through a two-speed and reverse gearbox and a de Dion-type transmission. The blades of both turbines are investment castings in a heat-resistant alloy developed by General Motors ; the nozzle rings of another special alloy are cast in shell moulds. Starting is by a 24 V motor and the light-up occurs at 3,000 r.p.m. while idling speed is 8,000 r.p.m. The exhaust temperature is about 538° C. Scoops arranged below the fins and forward of the rear wheels pick up air for an oil cooler and to insulate the exhaust pipe. Fuel is carried in a 35-gallon-capacity plastics tank mounted in the nose of the body.

Assembled as a unit with the transmission and installed immediately behind the driver, the maximum output is 370 b.h.p. and the weight 775 lb ; or specifically 2·1 lb/b.h.p. Space occupied is 63 in long, 32 in wide and 26 in deep. The complete vehicle weighs 2,800 lb, giving a weight-to-power ratio of 7·5 lb/b.h.p. It has been run at the General Motors proving ground and elsewhere but no figures of performance in these early tests were disclosed.

The Chrysler experimental turbine unit incorporates a heat exchanger and is claimed to have a fuel consumption comparable to that of an equivalent piston engine. This statement is qualified by the assumption that 120 s.h.p. (the output of the turbine unit) is equal to 160 s.h.p. in a piston engine due to the turbine's superior torque characteristics. The unit comprises a centrifugal compressor, driven by a single-stage turbine, delivering air through the heat exchanger to a single combustion chamber. The independent single-stage power turbine, in series with the compressor turbine, drives the output shaft through a 2-stage reduction gear. After passing through the heat exchanger the exhaust is discharged from an elliptical tail pipe, 10 in wide, arranged in the conventional position below the boot of the car.

With the heat exchanger mounted immediately above the compressor-turbine set, the complete power unit measures 32 in long, 33 in wide and 28 in high, and was installed under the bonnet of a standard production Plymouth Belvedere sports coupé. It is stated to be 200 lb lighter than the standard piston engine and gear unit : the only transmission gear required for the turbine being a simple reverse gear. Any fuel from gasoline to fuel oil can be used and, in the Plymouth car, the consumption was reported to be at the rate of 15 to 20 m.p.g.

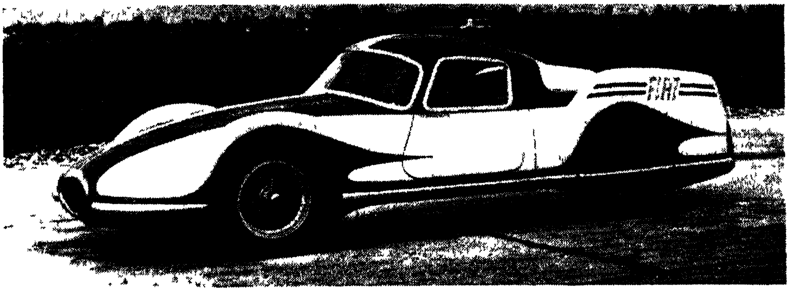


Fig. 16 : 24—The turbine unit of the experimental Fiat car develops 200 b.h.p. at 22,000 r.p.m.

In August 1954 Austin announced that a 125 h.p. gas turbine unit fitted with a heat exchanger had been successfully tested in an Austin Sheerline saloon. A heat exchanger is held to be an essential component to enable the rate of fuel consumption to be brought within acceptable limits. The complete unit comprises a two-stage compressor driven by a three-stage turbine and followed by an independent power turbine. Exhaust gas is passed through the cross-flow exchanger to transfer its heat to the air from the compressor before it reaches the combustion zone.

Another experimental vehicle was exhibited by Fiat at the Turin Show in April 1954. This was a two-seater coupé with the power unit mounted

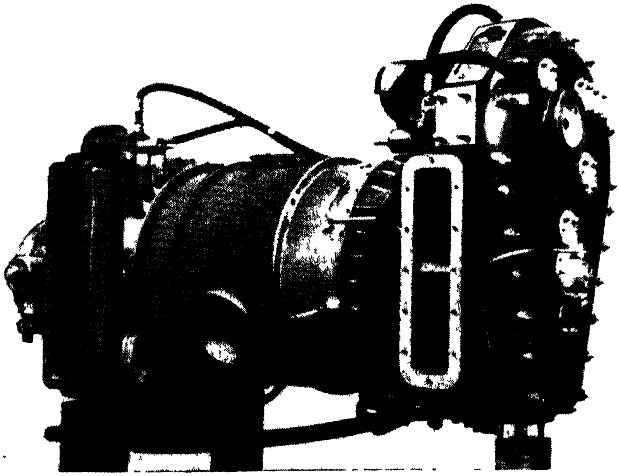


Fig. 16 : 25—Turmo two-shaft turbine unit for tank propulsion.

at the rear and fed from an air intake at the nose. Of the relatively high-compression type it has a two-stage centrifugal compressor driven by a two-stage turbine. Gas temperature at the turbine nozzle is about 800° C. The single-stage independent power turbine drives through reduction gears to a de Dion rear axle unit and no change speed gears are employed. Control is by two pedals only—accelerator and brake. It is understood to develop about 200 b.h.p. with the compressor rotating at 22,000 r.p.m. No heat exchanger is provided and the fuel consumption is at a rate approximately twice that of a piston engine of comparable output.

The chassis is of straightforward tubular construction with all wheels independently sprung on wishbones and coil springs. Apart from exceptionally large area fins above the rear wings the body, evolved after wind tunnel tests, is unexceptional. The large-diameter exhaust duct is directed horizontally to the rear. Immediately prior to the Show it was demonstrated at the Turin airport and reached a speed of 135 m.p.h. but it is expected to attain 155 to 160 m.p.h. after further development.

The Société Turboméca has developed a two-shaft unit specifically for the propulsion of military tanks and for marine craft. Designated the Turmo, this is basically an Artouste I charging set followed by an independent turbine driving through a compound planetary gear which reduces the speed at the output shaft to a maximum of 3,500 r.p m. The air mass flow is 4·75 lb/sec and the pressure ratio 3·8 : 1. Maximum output is 270 b.h.p. with a fuel consumption of 1·0 lb/b.h.p./hr but the rating for continuous operation is 235 b.h.p. with consumption at 1·08 lb/b.h.p./hr. Overall dimensions are 45·1 in long, 21·6 in wide and 27·2 in high, and the weight (without reduction gear) is 300 lb.

CHAPTER 17

Other Turbine Applications

So rapid and widespread has been the adoption of the gas turbine for aircraft propulsion that its potentialities for other power applications have been somewhat obscured. Actually, the earliest gas turbines were conceived as stationary power units but failed to establish themselves owing to insufficient knowledge of principles, poor component efficiencies and inadequate constructional metals. The aircraft turbine inherited these handicaps, of course, but they were more than offset by operation at altitude in ambient atmospheric conditions of lower temperature and density and by the benefit conferred by ram compression when travelling at high speed. In the case of the turbojet, the method of propulsion by direct reaction simplified the power unit as only a single-stage turbine was necessary to drive the compressor and no gearing was required.

Apart from many speculative proposals, scarcely capable of realization at the level of technical development then obtaining, the Stolze hot-air turbine project of 1872, with axial air compressor, gas producer, heat-exchanger, combustion chamber and axial turbine, represents the practical conception of the modern constant-pressure cycle plant. In 1906 the Swiss firm of Brown-Boveri designed and built the first multi-stage centrifugal compressor for a constant-pressure gas turbine built by Lemale and Armangaud in Paris.

About this period a long-sustained effort to develop the constant-volume gas turbine was commenced by Holzwarth in Germany. In this type several combustion chambers arranged around the turbine shaft were charged sequentially with fuel and air at about 20 lb/in^2 abs through separate valves. After ignition, the pressure rise in the chamber opened a spring-loaded valve and the combustion gases passed to a nozzle in which they were expanded and acquired velocity before impinging on the rotor blades. When the pressure in the chamber had fallen, scavenge air was passed through to clear the chamber ready for the next charge.

Brown-Boveri built the first Holzwarth turbine about 1910 but it was not a success. Subsequently several units were built in various sizes up to 5,000 kW (6,700 h.p.) capacity. Tests of a 1,700 kW (2,280 h.p.) set in 1923 gave an overall thermal efficiency of 14·2 per cent. As late as 1930 a 2,000 kW (2,680 h.p.) set with water-cooled rotor and blades was built and tested by Brown-Boveri.

The constant-volume cycle compares unfavourably with the constant-pressure cycle since operation is intermittent, the gas flow from the individual chambers is of varying velocity and consequently blade angle and speed must be a compromise, the valve gear is a complication and the valve porting is inimical to a high mass flow. The system has not gained ground and does not challenge the more conventional constant-pressure cycle.

Between the wars there was continued development of a specialized type of gas turbine, the exhaust turbo-supercharger for aircraft petrol engines and marine and stationary diesel engines. Rotary compressors of all types also received attention and the work in these two spheres furnished a background of designing and operating experience which was drawn upon by the designers of the early aircraft power units.

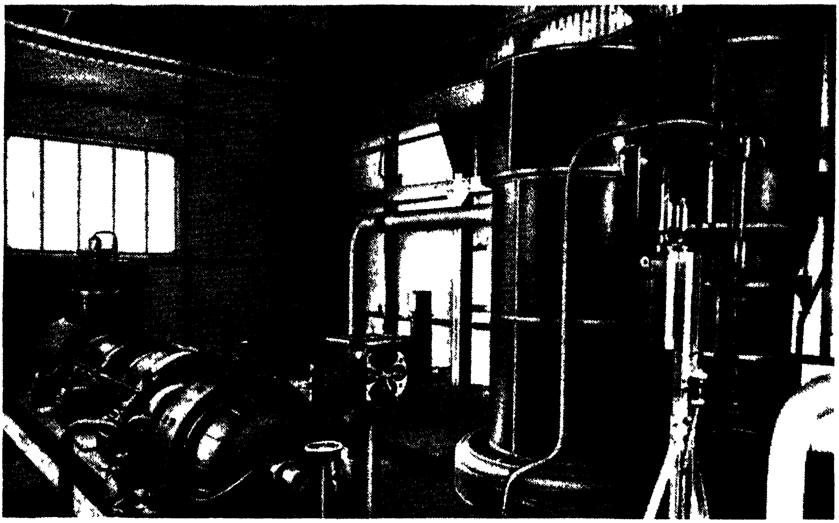


Fig. 17 : I—Gas-turbine-driven compressor for Brown-Boveri Velox quick-steaming boiler.

Exceptionally, the direct development of gas turbines for industry was pursued by Switzerland, and in particular by the Brown-Boveri Co. To that concern must be given the credit for the first successful applications of the gas turbine for commercial operation.

EARLY COMMERCIAL APPLICATIONS

From 1932 onwards Brown-Boveri has used gas turbines, running on blast furnace gas, natural gas or heavy oil, in an auxiliary capacity to drive the combustion air compressors of their Velox quick-steaming boilers. These units operate with gas temperatures of 600° to 650° C at the turbine nozzle and give reliable service in conformity with the standards of the steam plant.

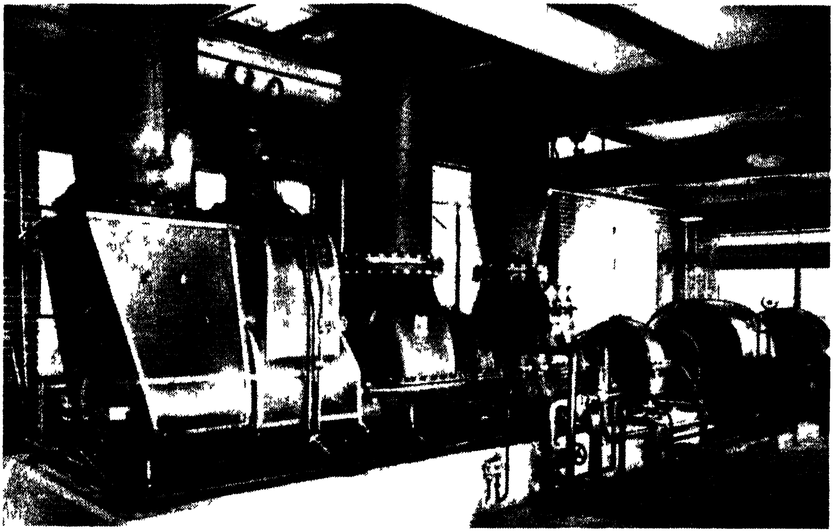


Fig 17 2—Turbo-compressor set for Houdry catalytic cracking plant

For the catalytic cracking of petroleum for the manufacture of high-octane petrol by the Houdry process it is necessary to furnish large quantities of air at relatively high pressure to regenerate the catalyst. A typical plant handling 10,000 barrels per day requires air at 45 lb^2 pressure at the rate of 40,000 ft^3/min. The Sun Oil Co , Philadelphia, in 1936 installed the first Brown-Boveri turbo-compressor set for this duty

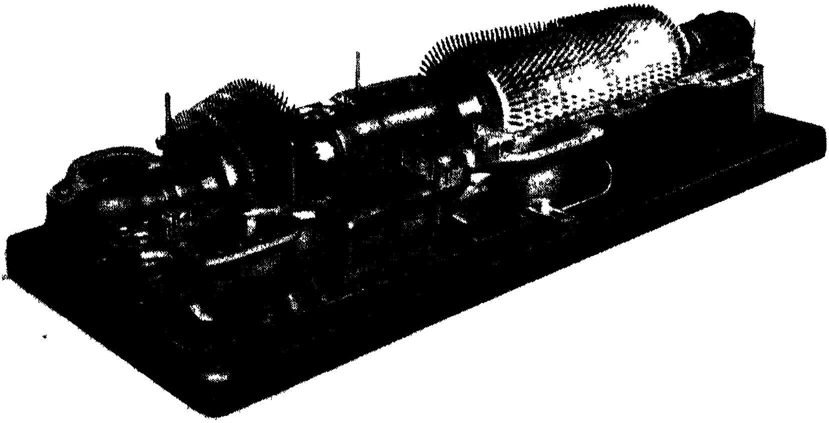


Fig. 17 : 3—Five-stage turbine and 20-stage compressor unit of Houdry set.

The air burns the carbon deposit from the catalyst and the products of combustion are delivered at a temperature of 500° to 550° C and pressure of 40 lb/in^2 to drive the five-stage turbine. This unit develops about 6,200 h.p. and drives the 20-stage axial compressor delivering air to the catalyst chamber. Operating speed is 5,180 r.p.m.

Economic efficiency of the process is secured only by using the products of regeneration to provide the energy required to compress the air. Actually the turbine produces more power than is necessary to drive the compressor and the surplus is utilized to generate up to 900 kW of electric power and also steam for use in other phases of the process. In 13½ years' operation this first unit lost only 43 days out of 4,345 possible working days, or 1·0 per cent, for maintenance and repairs. More than 30 similar units built by Brown-Boveri and their licensees, the American firm of Allis-Chalmers, have since been installed and are, in fact, an essential feature of the process.

GENERATING SETS—OPEN CYCLE

In 1939 the first gas turbine generating plant designed for a specific commercial application was exhibited and run at Zurich by Brown-Boveri. Subsequently it was installed in an underground power station at Neuchâtel and has served since 1940 as a stand-by and peak-load plant. For this duty it was designed on the simplest and most economical lines as regards first cost, operation and maintenance. It comprises a compressor, combustion chamber, turbine and generator only, no heat exchanger being provided. Diesel oil is used as the fuel and the output is 4,000 kW. The set is run up at regular intervals for an operational check and is always available for service. It is started up automatically by a diesel engine when a failure occurs in the power supply system. Normally full load is taken up 12 min after starting from cold, but in emergency it can be carried after 8 min.

Another 4,000 kW plant of simple type without heat exchanger was put into service at a power station at Chimbote, Peru, in 1949.

Two 1,650 kW simple-cycle sets with heat exchangers were supplied by Brown-Boveri in 1949 as the sole source of power for a cement works in Venezuela. Operating on a mixture of 60 per cent heavy bunker oil and 40 per cent diesel oil, the average rate of fuel consumption was 1·08 to 1·14 lb/kW/hr. Regular maintenance and cleaning was undertaken to ensure absolute reliability and continuous production in the factory. The residual fuel had a high ash content and deposits on the turbine blades were washed off, by built-in equipment, after 900 hr operation. After the first of the two larger natural gas units, mentioned later, was put into operation at the plant in November 1952, these two smaller sets were changed over to natural gas firing and the periodic cleaning of the turbines became unnecessary. Cement dust which penetrates the

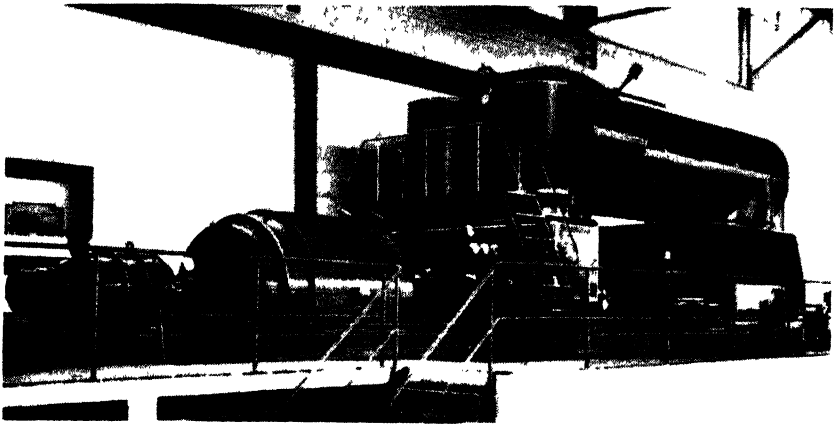


Fig. 17 : 4—Brown-Boveri 4,000 kW stand-by plant for Neuchâtel.

air filters and builds up on the compressor blades is washed off after 3,000 hr operation. Heat exchangers are cleaned in service by built-in soot blowers taking compressed air from the turbine set. The plant is shut down each week on Sunday and each set runs over 4,000 hr per annum. Starting power is supplied by a diesel engine which also serves to generate about 250 kW for essential services at the week-end.

The largest generating plant yet built is the Beznau station of the Nord-Ostschweizerische Kraftwerke in Switzerland. This power supply company distributes the output of four water-power plants and the thermal station is required to take up load, particularly in the winter months, when a scarcity of water prevails. Two Brown-Boveri compound-cycle sets are installed. One 13,000 kW set was put into operation in 1948 and the second, rated at 27,000 kW, in the following year. These outputs are based on winter operation with an average ambient temperature of 5° C (41° F). Both sets are of similar design with a low-pressure system in which the l-p turbine drives l-p and m-p compressors at a speed varying with the load and a high-pressure system with the h-p turbine driving the h-p compressor and the alternator at constant speed.

After passing through the l-p, m-p and h-p compressors, with intercoolers between the three units, the air is preheated in the heat exchanger and then raised to a temperature of 600° C in the first combustion chamber. The efflux from the h-p turbine is reheated in a second combustion chamber before reaching the l-p turbine which exhausts through the heat exchanger

Grade III boiler oil was at first used as fuel but later a change to Grade IV oil was made. Some ash is deposited on the turbine blades and must be washed off periodically, after several hundred hours operation. This process is quickly effected and is usually performed before starting up the set after the week-end shut-down. Soot also tends to accumulate in the heat exchanger and thus presents a fire hazard. This is prevented by soot-blowing every second day and a wash-out after a prolonged period of operation.

The operational difficulties were eradicated as they arose and with the experience gained in two winter seasons some modifications were effected. Both sets are now giving reliable service at an overall efficiency exceeding 30 per cent.

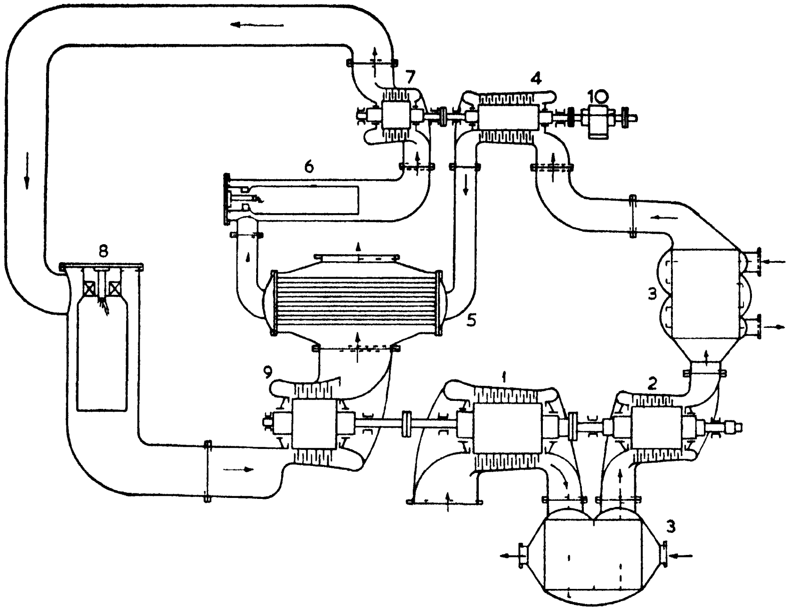
In 1949 a set dimensionally similar to the 13,000 kW set was installed at the Santa Rosa plant at Lima, Peru, also for service in conjunction with hydro-electric generators. On this site the ambient temperature ranges between 15° and 25° C and the set is rated at 10,000 kW. Diesel oil is used as the fuel and no fouling of the turbine has occurred. Operating over a period of 8,332 hours up to the end of 1951 this set generated 60·5 million kWh with a total fuel consumption of 21,600 metric tons. This corresponds to a specific consumption of 0·786 lb/kW/hr and represents an overall efficiency of 24 per cent when operating at only 73 per cent of full load.

In Britain the first gas turbine generating set to be installed for commercial operation was the 15,000 kW set for the British Electricity Authority, at Stretford, near Manchester. This was built by the Metropolitan-Vickers Electrical Co. Ltd, and was first put on load in August 1952. Of the compound open-cycle type, the overall compression ratio is about 10 : 1 and the gas temperature at the turbine nozzles is 650° C.

Air is drawn through an electrostatic filter room and flows through a 14-stage l-p compressor, intercooler, 15-stage h-p compressor, and a heat exchanger to reach the h-p combustion chamber. The fuel used initially was gas oil but a change to a heavier grade will be made when sufficient operating experience has been accumulated. From the combustion chamber the gases are delivered to the two-stage h-p turbine, which drives the h-p compressor, and thence to the l-p combustion chamber where the gases are reheated before admission to the six-stage l-p turbine driving the l-p compressor and the alternator. The effluent passes through the heat exchanger and is exhausted by way of a tall chimney stack.

The h-p units run at varying speed according to load up to a maximum of 5,500 r.p.m. but the l-p set and the alternator are maintained at constant speed of 3,000 r.p.m. Starting is effected by a 400 h.p. electric motor which rotates the h-p set up to 3,000 r.p.m., its idling speed.

An impression of the magnitude of these main generating sets may be obtained from particulars of some of the Metrovick component units. For instance, the intercooler has four stacks of water tubes, each stack comprising 232 tubes, about 12 ft in length. The total weight is 27 tons.



1 L p compressor 2 M p compressor 3 Intercooler 4 H p compressor 5 Heat exchanger
6 H p combustion chamber 7 H p turbine 8 L p combustion chamber 9 L p turbine
10 Reduction gear

Fig 17 5 Schematic layout of Brown-Boveri compound-cycle set.

Fig. 17 . 6—Beznau power station with two Brown-Boveri compound-cycle sets developing a total of 40,000 kW.

For the heat exchanger, which has a total tube surface area of 66,100 ft^2, there are six units each embodying 2,005 tubes about 21 ft long. Each of these units weighs 23 tons.

A 1,000 kW alternator set for auxiliary service in the Navy has been developed by W. H. Allen, Sons & Co. Ltd. of Bedford. The design was influenced by stipulations regarding space occupied, and performance and control were strictly specified to permit operation in parallel with existing steam turbo-alternators. Overall dimensions, excluding the alternator, are height 7 ft, width 8 ft 6 in and length 16 ft and the weight is 8·5 tons. The fuel is Admiralty diesel oil, B.310.

Air is trunked to the 13-stage axial compressor, which has a pressure ratio of 4·25 : 1, and then passed by way of 16 diffuser tubes to the annular two-pass, cross-flow heat exchanger enshrouding the drive shaft. Returning from the exchanger it is delivered to eight tubular combustion chambers lying parallel around the turbine cylinders. The flow direction is again reversed for the gases to reach the nozzle of the two-stage turbine which drives the compressor through a flexible coupling. Running at a speed of 8,000 r.p.m., the air mass flow is 37 lb/sec and the maximum gas temperature at the nozzle is 650° C under temperate or 700° C under tropical conditions.

The single-stage, low-pressure power turbine is independent of the compressor set and runs at 6,750 r.p.m. Output is transmitted by a tubular shaft passing through the heat exchanger to an Allen-Stoeckicht epicyclic gear that reduces the speed to 1,500 r.p.m. for the alternator.

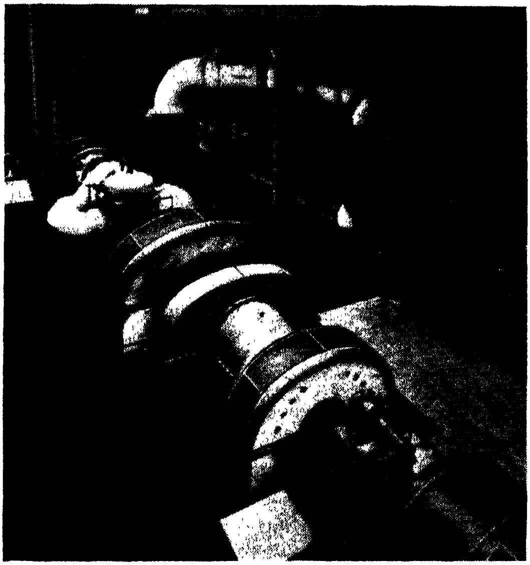
Fuel consumption is at the rate of 0·74 lb/b.h.p./hr or about 0·98 lb/kW/hr. Starting is by an electric motor which runs the compressor set up to 1,500 r.p.m. and after lighting-up assists the turbine to a speed of 2,500 r.p.m. The time required is about 1 min and normal procedure is to run at no load for 5 min to warm up. If necessary, however, the set can be started and brought on to full load in approximately 1½ min.

The range of gas turbines to be built by this company includes 1,000 kW to 5,000 kW base-load or peak-load generating sets of 30 per cent thermal efficiency and simple, lightweight 350 kW generating sets and 200 h.p. units.

Adaptable for a wide range of stationary or transportable generating sets or for industrial duties, the 900 kW Ruston gas turbine is now well-proved and is in series production. Designed as a ‘ long-life ’ turbine, the first unit was installed in one of the firms’ power stations in 1949. The component elements are self-contained and self-supporting and thus the set can be readily dissembled into convenient units for transportation. Maintenance is also facilitated. To meet varied installation requirements all the ducting connections of components are arranged to adjust by steps of 15° through 360°.

In its simplest form, without recuperation, the 13-stage compressor

Fig. 17 : 7 — Metrovick 15,000 kW set at Trafford power station, Stretford, near Manchester.



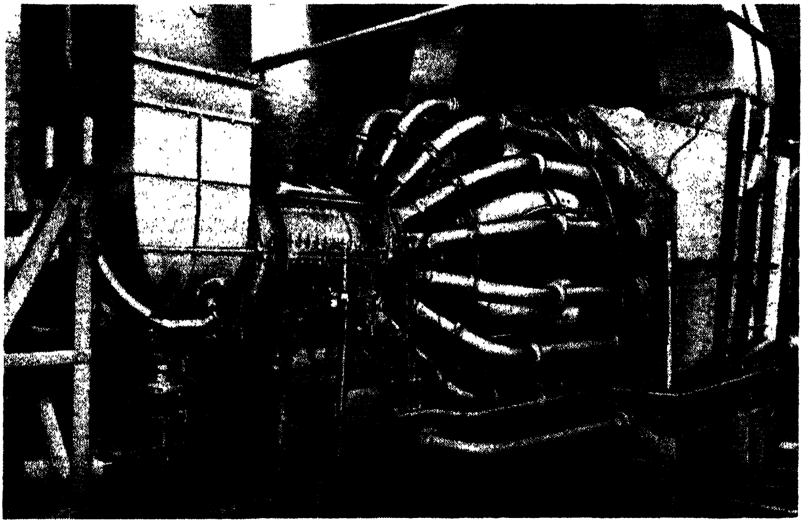


Fig. 17 : 8—Allen 1,000 kW auxiliary generator set for naval service.

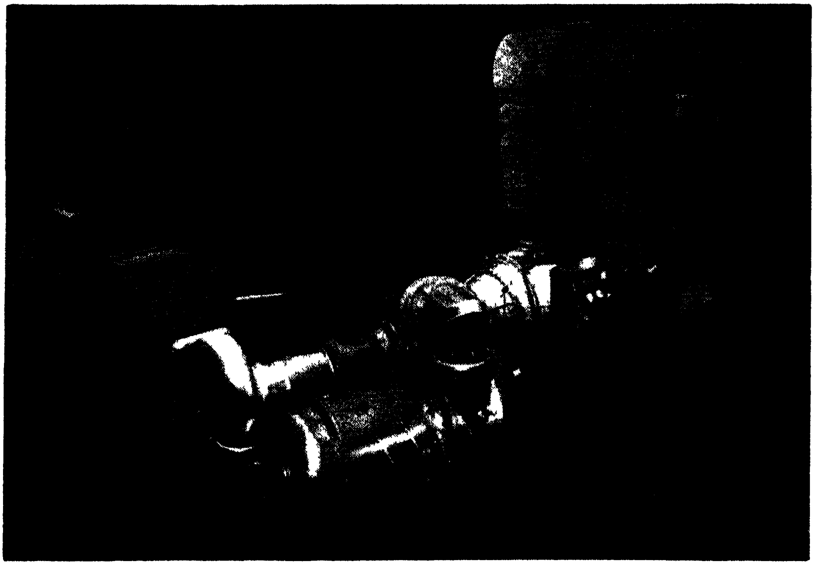


Fig. 17 : 9—Ruston 900 kW non-recuperative set on test.

delivers air at a pressure of 58·8 lb/in^2 abs at the rate of 22·6 lb/sec to a single, reverse-flow combustion chamber fitted with a Shell-type, louvred-wall liner. The compressor is driven by a two-stage turbine and an independent two-stage turbine in series provides the useful power output through an Allen-Stoeckicht epicyclic reduction gear. Air tapped from the main compressor is used to cool the turbines. From the eighth stage to the h-p intake, the h-p and l-p stators and the four faces of the l-p discs; from the 11th stage by way of the rotor axis to the inner and rear faces of the h-p discs ; and from the final stage to the front face of the first h-p disc. Turbine blades are of Nimonic 80A with fir-tree root fixings. Maximum speeds are 11,500 r.p.m. and 6,000 r.p.m. for charging and work turbines respectively and the reduction ratio can be arranged to give either 1,500 r.p.m. or 1,800 r.p.m. for 50-cycle or 60-cycle alternators. With a maximum gas temperature at the turbine nozzle of 727° C the output is 1,260 h.p. or an electrical output of 900 kW.

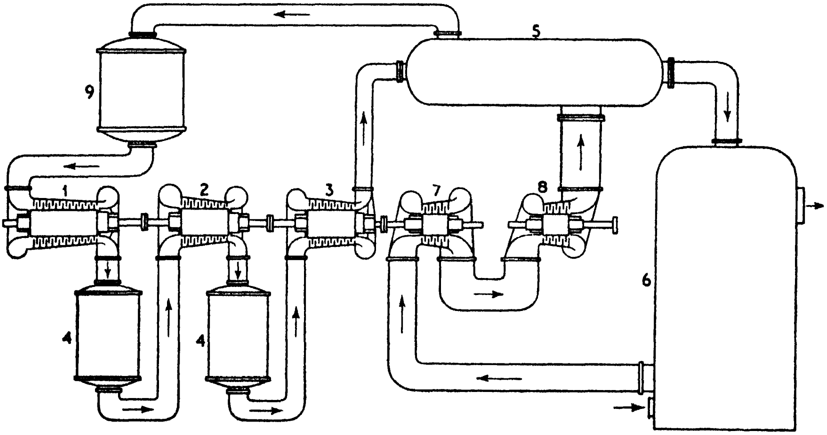
The inclusion of a heat exchanger in the cycle reduces the air mass flow to 21·4 lb/sec and lowers the output to 1,050 h.p. or to 750 kW but raises the cycle efficiency from 15·9 per cent to 22·5 per cent. Starting is by a 15 h.p. electric motor or a compressed air motor. With minor adjustments of the combustion system the fuel may be natural gas, methane, distillate oil, residual oil or a coal-tar fuel. Eight sets to run on sewage gas are being built for the L.C.C. sewage works at Beckton, two for operation on coal-tar fuel for the Bank of England and the Crittall Manufacturing Co. and one on natural gas for Italy.

CLOSED CYCLE SYSTEM

All the sets referred to previously are of the open-cycle type, with a continuous full intake of air and discharge of effluent. Another Swiss firm, Escher Wyss Engineering Works Ltd. of Zurich, has developed the closed-cycle type in which air, or some other gaseous medium, is continuously circulated and heated indirectly in a so-called ' air boiler '. The pioneer 2,000 kW set in the firm's works has been run experimentally since the summer of 1939 and has done periods of service feeding electric power into the public supply system.

The working medium in the closed circuit, in this instance air at 88 lb/in^2 abs and 21° C, is compressed in l-p, m-p and h-p compressors with intervening coolers and delivered at 352 lb/in^2 abs and 63° C to the heat exchanger. Here the temperature is raised to 350° C and it then flows to the air boiler and is further raised to 650–700° C by the external combustion of oil fuel.

This hot air is expanded through the h-p turbine driving the three compressors at 8,000 r.p.m. and through the independent l-p turbine driving the generator at 3,000 r.p.m. The output of the h-p turbine is 3,000 kW, representing 60 per cent of the total power developed. Leaving the l-p turbine at 88 lb/in^2 abs and about 400° C the air gives up heat in passing through the heat exchanger and finally the precooler before being recirculated.



1. L-p compressor: 2. M-p compressor: 3. H-p compressor: 4. Intercoolers: 5. Heat exchanger: 6. Air boiler: 7. H-p turbine: 8. L-p turbine: 9. Precooler.

Fig. 17 : 10—Schematic layout of Escher Wyss 2,000 kW closed-cycle turbine plant.

Working under ideal conditions with no loss, it can be shown that for specified maximum and minimum temperatures of the medium, the thermal efficiency depends solely on the ratio of the maximum and minimum pressures and not on the absolute magnitude of the pressures. The lower the pressure ratio the higher the efficiency. In practice, of course, heat is lost at the compressor intercoolers, at the precooler, at the furnace flue, by friction and radiation. Nevertheless, the principle holds good approximately for a system in which losses occur. Whilst a steam plant requires high pressures and high temperatures to achieve efficiency, in the closed-circuit gas turbine plant it is not necessary to raise pressures to improve efficiency. Thus it offers the possibility of substantial advantages in weight, first cost, and maintenance of the main components and also the control equipment.

The separation of the gas circuit from the combustion process enables the flow passages to be of reduced cross-section, as there is no likelihood of impedance or restriction by accumulation of solid matter from the products of combustion. Size and weight are also favourably influenced owing to the entire circuit operating under pressure, as the rate of heat transfer is nearly proportional to the pressure.

With an initial pressure of 10 atm abs at the compressor intake, instead of the 1 atm abs as is usual with the open-cycle system, and with equal temperatures and speeds, the diameters of both turbine and compressor could be reduced by 66 per cent.

Also of importance is the facility for control by regulating the density of the working medium in the closed-cycle without altering either the temperature or the speed. If, in the smaller units referred to, the pressure at the compressor inlet is lowered from 10 atm abs to 1 atm abs, the output would be reduced by 90 per cent. Actually the medium in the closed cycle can be reduced to a negative pressure in order to reduce the power required for initial rotation when starting up the plant.

In the first Escher Wyss plant the pressure ratio was approximately 4 : 1. An indication of the small size of the working units of a closed-cycle set, as compared with similar units for an open-cycle set operating at atmospheric pressure, is given by the dimensions. The mean external diameter of the h-p turbine rotor is 13·75 in while the diameters of the rotors for the compressors range from 10·6 in to 15·0 in.

Using in the air boiler a fuel oil having a minimum calorific value of 18,450 B.Th.U./lb, the specific fuel consumption of the complete set is, at full load, 0·614 lb/kW/hr. The thermal efficiency of the turbine plant —with all auxiliaries but excluding the electric generator—is 31·6 per cent at full load. It is of interest that the efficiency curve falls very slowly as the load is reduced. At half-load it is 29·5 per cent and even at one-fifth load it has fallen only to 24·4 per cent.

The external firing of a closed-cycle system, of course, makes possible

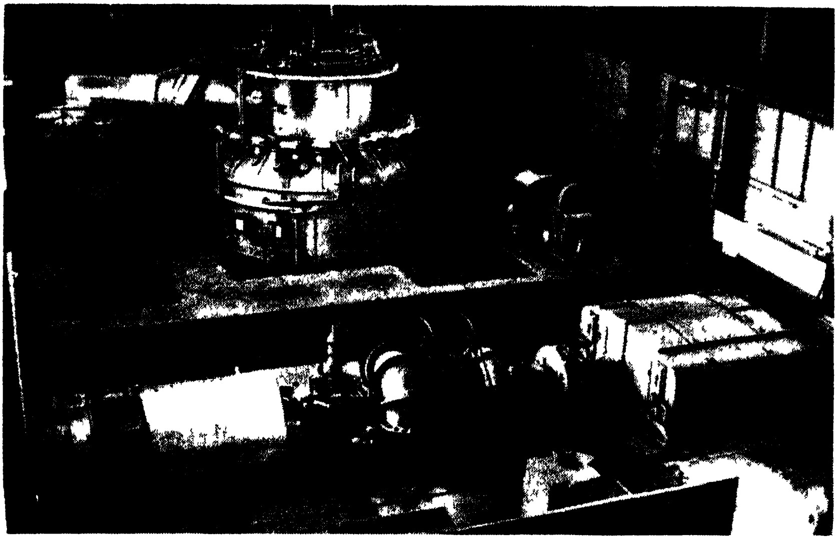


Fig. 17 : 11 Escher Wyss 12,500 kW closed-cycle set at St. Denis, Paris.

the use of cheap gaseous, liquid or solid fuel without contamination of the working medium or damage to the rotating components.

The first plant for commercial operation was the 12,500 kW set built by Escher Wyss for Electricité de France and installed at St. Denis, Paris. Its overall efficiency is 34 per cent.

In Britain, John Brown and Co. Ltd. hold a licence to manufacture closed-cycle plants on the Escher Wyss system, known as the AK system after its progenitors Professor Ackeret and Dr. Keller. A 12,500 kW set has been built for the North of Scotland Hydro-Electric Board and installed near Dundee. In general characteristics it resembles the St. Denis set and has a layout slightly modified from the original.

The l-p turbine drives the l-p compressor and the generator while the two i-p compressors and the h-p compressor are driven by the h-p turbine. Three intercoolers are provided between the compressors. The efflux from the h-p turbine is reheated in the air boiler before admission to the l-p turbine, from which it passes through the main heat exchanger and a precooler before being recirculated. For the air boiler a separate open-cycle turbo-system is employed. A turbine running on the flue gas from the boiler drives the combustion air compressor and also a blower which recirculates a portion of the flue gases through the boiler. After compression the combustion air is preheated in an exchanger through which passes the turbine efflux.

Studies of closed-cycle turbine plants using a gas as the working medium have been made by the English Electric Co. Air and vapours, for example steam or mercury vapour, were specifically excluded. Since there is no loss of the medium in the closed circuit it is possible to employ relatively rare gases. The outstanding feature of the system is the use of a monatomic gas as the medium.

A monatomic gas may be briefly defined as one in which the molecule, the smallest division retaining all the characteristics of the gas, contains only one atom. The molecules of diatomic and triatomic gases, carbon monoxide and carbon dioxide, for instance, have respectively two and three atoms. For a monatomic gas the ratio of the specific heat at constant pressure to the specific heat at constant volume is approximately 1·66 as compared with 1·4 for a diatomic gas and 1·3 for a triatomic gas. The higher the value of this ratio the lower will be the pressure ratio required for a compressor and turbine designed to give maximum efficiency at chosen values of maximum and minimum limits of temperature in the system.

Furthermore, within given temperature limits and with a given pressure ratio, the greater the density of the gas the smaller will be the adiabatic heat drop in the turbine. As a consequence, providing that rotor diameters, speeds of rotation and ratios of blade speed to gas speed remain the same, the number of stages required in the turbine and in the compressor will be fewer. Additionally, the greater the density of the gas the smaller will be the dimensions of the flow passages. A monatomic gas is not liable to dissociation at high temperatures, and if a relatively inert gas is used, the corrosive action on the turbine and compressor blades will be less than that of air.

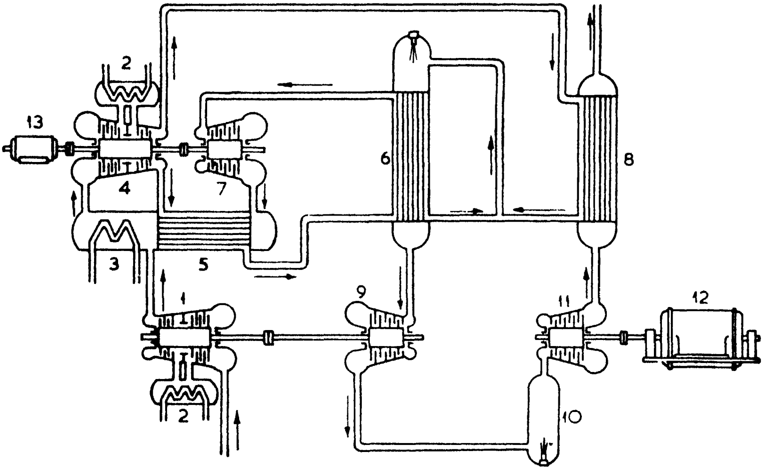
By using a monatomic gas, such as argon, all these advantages can be realized. This inert gas has a density approximately 1·38 times that of air at the same temperature and pressure. The rare gases krypton and xenon could also be used to further advantage, as their densities are respectively 2·87 times and 4·53 times that of air. Some proportion of neon, another monatomic gas, could be included in the working medium, although its low density, only 0·696 that of air, is a disadvantage.

The question of relative density is of considerable importance. As an alternative to the comparatively rare gases mentioned, the English Electric Co. propose the use of carbon dioxide for the working medium, as it has a density of 1·52 times that of air at the same temperature and pressure. This is despite the fact that it is a triatomic gas and consequently less suitable for the purpose than a monatomic gas.

MIXED-CYCLE SYSTEM

Differing from both open-cycle and closed-cycle types is the mixed-cycle, high-pressure system developed in Switzerland by Sulzer Bros. of Winterthur. By operating at a high density the rotating components can be relatively small and heaters and exchangers do not need to be as large as those for the closed system. There are two circuits, one indirectly heated and the other directly heated. The second circuit draws continuously h-p air from the first which is continuously made up by a l-p compressor.

In the first circuit, air at about 70 lb/in^2 abs is raised by the h-p compressor 4—two units with intercooler—to about 285 lb/in^2 abs. Part of this air flows through the heat exchanger 5 and the air heater 6 in which the temperature is raised to 650° C. It is then expanded down to 70 lb/in^2 abs in the air turbine 7, which drives the h-p compressor, and flows through heat exchanger 5 and precooler 3 for recirculation at 20–25° C by the compressor 4.



1. Make-up compressor: 2. Intercooler: 3. Precooler: 4. H-p compressor: 5. Heat exchanger: 6. Air heater: 7. Air turbine: 8. Heat exchanger: 9. H-p turbine: 10. Reheat chamber: 11. Output turbine: 12. Generator: 13. Starting motor.

Fig. 17 : 12—Diagrammatic layout of Sulzer mixed-cycle plant.

High-pressure air abstracted from the circuit is passed through heat exchanger 8 and delivered to the combustion chamber of air heater 6. The products of combustion are expanded to about 100 lb/in^2 abs in the h-p turbine 9, are restored to their initial temperature in reheat chamber 10 and finally expanded down to atmospheric pressure in the l-p turbine 11. They are discharged through heat exchanger 8 where they give up part of their residual heat to the combustion air.

To replace the air used for combustion in the open circuit, a l-p make-up compressor 1, comprising tandem units with intercooler, draws air from the atmosphere and delivers to the precooler 3 where it joins the circuit air. This compressor is driven by the h-p turbine while the l-p turbine provides the useful power output.

The air heater is not directly comparable with an air boiler for a closed-cycle system since only 50–60 per cent of the heat is transmitted through the tubes and the remainder is carried further by the combustion gases to the open circuit of the system. Another important aspect is that the tubes are subjected to the same high pressure both inside and outside. The specific heat transfer is thereby increased and the tubes are relieved of the mechanical stresses that are undesirable at the high operating temperature.

A 7,500 h.p. marine set was built for initial experimental operation. Later a 20,000 kW set was built for the Nord-Ostschweizerische Kraftwerke and installed at Weinfelden. It operates as a ‘winter power station ’ similar to the Beznau plant already mentioned.

An interesting fact emerges in regard to a thermal station installed for operation in conjunction with hydro-electric plants. Even though it should not be operated at all it may yield an overall economy since, with the thermal station in reserve, the hydro stations can be kept in operation longer in the event of a delayed thaw by running the reservoirs to a lower level than would otherwise be practical. Expressed otherwise, it is more economical to install a standby thermal station than to increase the capacity of the reservoirs.

NATURAL GAS SETS

Where natural gas is available relatively simple and inexpensive gas turbine plants for power generation are possible. An outstanding example occurs in the three Brown-Boveri 4,000 kW sets built for the Anglo-Iranian Oil Co. Ltd. Two are already installed at Tembi, Southern Iran. Since the fuel cost is very low and consequently a high thermal efficiency is of little import, these sets are of the simplest single-shaft type without preheating of the air. The ambient temperature at the site varies between 49° C and 20° C and the effective output varies accordingly from 3,250 kW to 4,700 kW. Operating speed is 3,800 r.p.m. which is reduced by gearing to 1,500 r.p.m. for the alternator.

Consisting mainly of gases such as methane, ethane and propane, the fuel has a high calorific value and is tapped at a natural pressure of nearly 400 lb/in^2 abs and a temperature of 21° C. Instead of the usual electric starting system a small expansion turbine driven by the gas at natural pressure accelerates the main unit to about 30 per cent of its normal running speed. During this period the gas is exhausted directly to atmosphere. This auxiliary turbine is permanently coupled through reduction gearing and in normal running serves to expand the gas fuel, which is preheated by the exhaust gas to about 300° C, down to a pressure of 64 lb/in^2 abs. In performing that function it develops about 150 kW which is transmitted to the main shaft.

Six basically similar sets are built or building for the Arabian

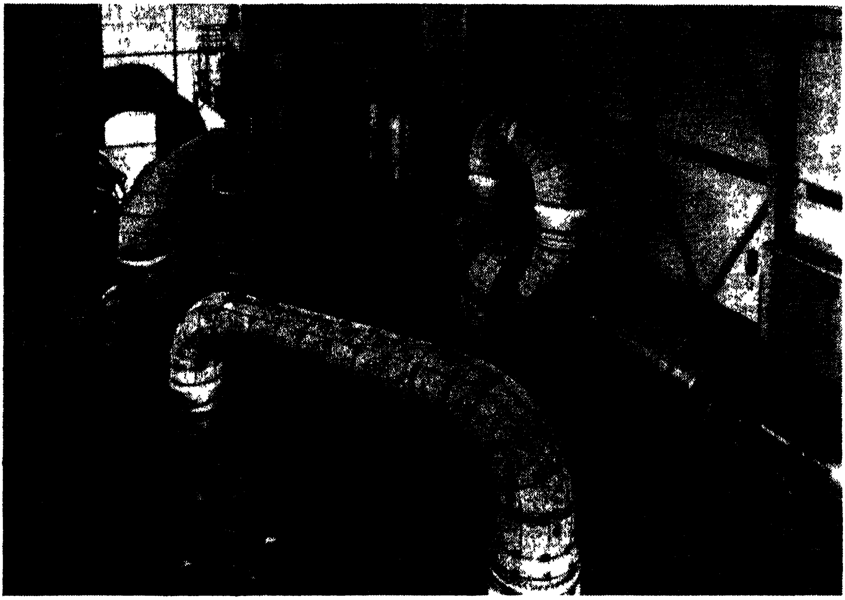


Fig 17 . 13 Sulzer 20,000 kW mixed-cycle plant at Weinfelden.

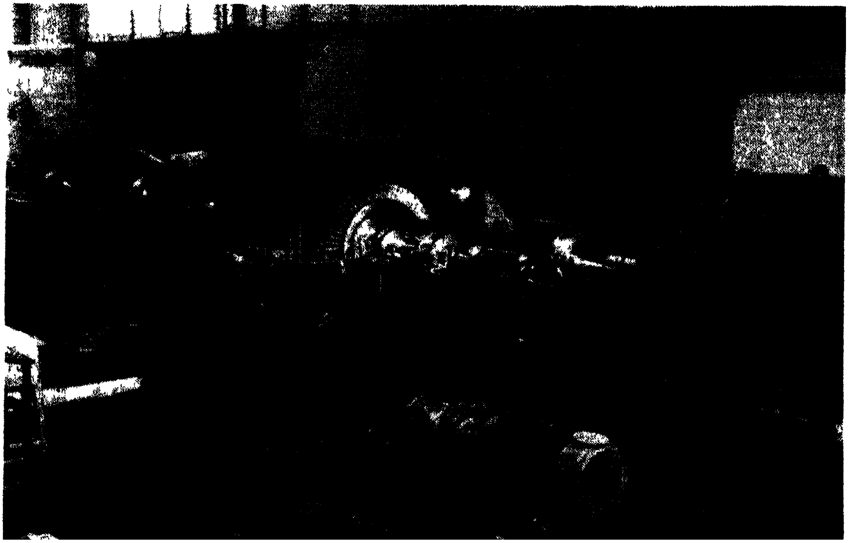


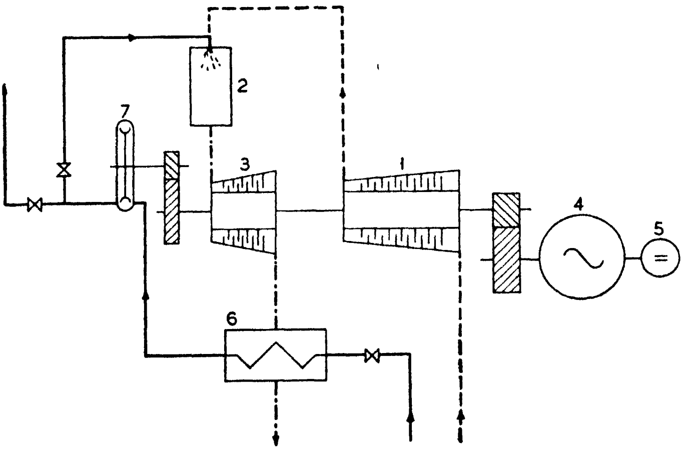
Fig. 17: 14—Brown-Boveri 4,000 kW natural gas set for Anglo-Iranian Oil Co.

American Oil Co. (ARAMCO). As a result of the satisfactory operation of the two 1,650 kW oil-fired sets mentioned earlier, the C. A. Venezolana de Cementos ordered two further sets of 5,000 kW each to run on natural gas. A 10,000 kW compound-cycle set without heat exchanger to run on natural gas was built by Brown-Boveri for the City Electricity Supply at Bucharest, Roumania. Although completed and test-run in 1946 it could not, due to political conditions, be installed or taken into service until 1951. It is now running at rated capacity for about 16 hr daily.

It is of interest to note that although these natural gas sets are erected and run at the maker's works, full trials are conducted on site after installation since natural gas is not available in Switzerland.

Britain has no substantial supplies of natural gas but it is hoped that a source of cheap fuel—which hitherto has been run to waste—can be exploited by means of the gas turbine. Reference is made to the ' firedamp ' or methane present in coal mines. This is drawn off in the upcast air at very low concentration ; probably less than 1·0 per cent. Providing it is compressed and adequately preheated, which will possibly necessitate the use of a supplementary fuel, it can furnish combustion gas for turbines capable of generating all the power required by the colliery. As a first experiment the Ministry of Fuel and Power and the National Coal Board are co-operating to install such a plant. The English Electric Co. is building the gas turbine.

Mention has already been made of the Ruston and Hornsby sets to run



——— Air ——— Fuel gas —·—·— Combustion gas

1. Compressor: 2. Combustion chamber: 3. Turbine: 4. Alternator: 5. Exciter: 6. Gas preheater: 7. Auxiliary turbine.

Fig. 17 : 15—Layout of natural gas turbine sets at Tembi, Southern Iran.

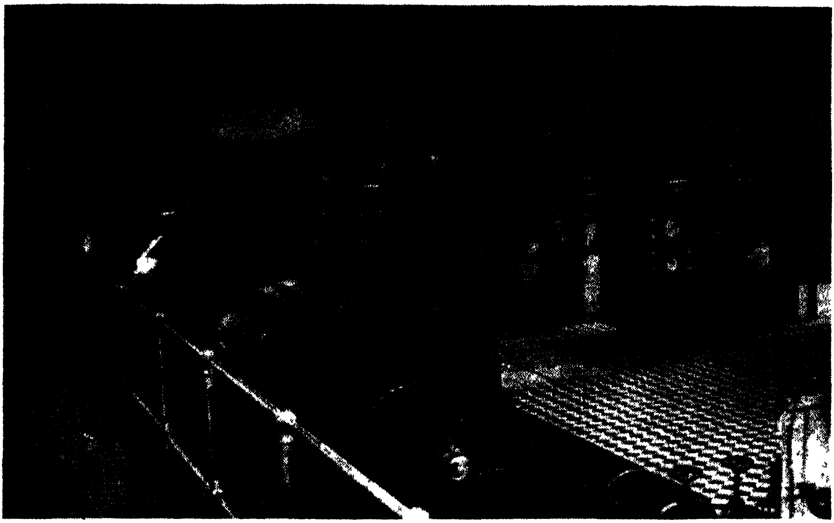


Fig. 17 : 16—5,400 kW set running on blast-furnace gas at the Dudelange steelworks.

on sewage gas and also on natural gas. Should the efforts now being made to effect the underground gasification of low-grade coal seams be successful the gas turbine could efficiently utilize the low calorific gas obtained.

FURNACE GAS

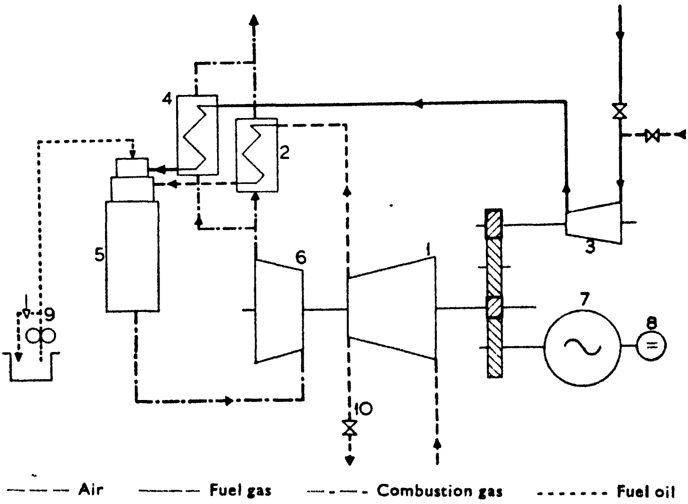
The gas turbine is likely to find increasing use in iron and steel plants as blast-furnace gas or coke-oven gas furnishes a relatively cheap fuel for generating electric power or driving blowers. A 5,400 kW set for that duty was built by Brown-Boveri for the Aciéries Réunies de Burbach-Esch-Dudelange in Luxemburg. It comprises a single-shaft turbine with air and gas preheaters and a fuel-gas compressor arranged as shown in the layout diagram. For periods when a supply of blast-furnace gas is not available—when starting up furnaces, for instance—the set can be run on oil fuel. Under such conditions the gas compressor, which is permanently coupled to the main unit, is changed over to supply additional air. Compressed air may be tapped off the main compressor for use in the steelworks or the blast furnace.

The calorific value of the gas varies, with a low value of about 1,300 B.Th.U./lb. Total consumption of the set is at the rate of 18·3 lb/sec, representing an overall thermal efficiency of about 21 per cent. The set can be operated to develop a maximum output of 5,400 kW, or up to 33 lb/sec of air can be taken off at 42·5 lb/in^2 abs for the steelworks or at 31·3 lb/in^2 abs for the blast furnaces. After 4,226 hr operation in 1951 the set was opened up and found in good condition with no heavy deposits. No replacements were necessary and it was immediately returned to service. By the end of 1952 it had been in operation for 10,000 hr and had produced 49·4 million kWh.

The first steelworks blower to be driven by a gas turbine running on blast-furnace gas was the 2,000 kW set installed by Brown-Boveri at Baracaldo, Spain, in 1951. Air for the turbine and the converter is compressed in a common axial compressor to 42·67 lb/in^2 abs. The blast air delivered amounts to 26,480 ft^3/min, or about 30 per cent of the total handled.

Two 7,500 kW open-cycle sets to run on blast-furnace gas are being built by Sulzer Bros. for the Usines Métallurgiques de Hainaut, at Couillet, Belgium. One drives an alternator only and the other an alternator and a furnace blower. Each set comprises l-p and h-p compressors with an intercooler, duplicate heat exchangers, a single combustion chamber and a turbine. Auxiliary l-p and h-p compressors, with an intercooler, are driven from the main set to raise the pressure of the blast-furnace gas before delivery to the combustion chamber. Operating with an air inlet temperature of 15° C and a cooling water temperature of 20° C the overall efficiency will be about 24 per cent.

John Brown & Co. Ltd. is building a 700 kW generator set for the Foleshill gasworks, Coventry. Of the closed-cycle type, it will utilize as fuel the flue gases from the continuous vertical retorts.



1. Compressor: 2. Air preheater: 3. Gas compressor: 4. Gas preheater: 5. Combustion chamber: 6. Turbine: 7. Alternator: 8. Exciter: 9. Oil fuel pump: 10. Air extraction.

Fig. 17 ; 17—Diagrammatic layout of 5,400 kW set to run on blast-furnace gas.

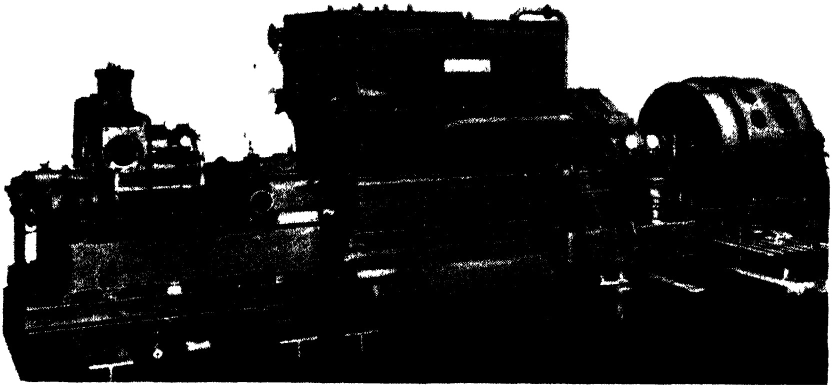


Fig. 17 : 18—A turbine fed with residual gases provides nearly all the power required for the compression of air in nitric acid manufacture.

INDUSTRIAL PROCESS GAS

In a number of chemical processes large quantities of compressed air are required and considerable heat is released. By utilizing the hot reaction gases in a gas turbine, power is generated to drive the compressor and make the process self-supporting. In certain cases, such as the oil-cracking process already mentioned, an excess of power can be obtained by this method while in others the energy recovered may fall short of requirements and a supplementary source of power must be provided. Nevertheless it represents a worth-while economy.

The production of nitric acid falls in this category. Air, raised to a pressure of about 130 lb/in^2 and preheated, and ammonia combine over a platinum catalyst in a reaction chamber. The reaction is exothermic and the hot gases containing oxides of nitrogen are passed through heat exchangers to a cooling tower where the nitric acid is formed and drawn off. Leaving the cooling tower, the cool residual gases, largely nitrogen, are reheated in the heat exchangers and then expanded through a turbine built of acid-resistant steel. The residual gases amount to about 80 per cent of the air supplied to the process.

In the pioneer Brown-Boveri set built in 1935 for the U.S.S.R., the multi-stage centrifugal air compressor with intercoolers required about 1,780 kW while the turbine developed some 1,400 kW. The necessary additional energy was supplied by an electric motor, so proportioned that in emergency it could supply the whole input to the compressor.

Several sets of this type have been supplied to Germany and the U.S.S.R. for nitric acid production. Five sets of 3,000 kW each running on hot air and ammonia gas were built for the Leuna synthetic fuel plant in Germany, the first in 1926.

PEAT FUEL

Another relatively cheap fuel available in certain areas is peat. As is the case when peat is burned under boilers in steam plant, the first problem is to remove the water. Air drying is too slow and cumbersome a process but with a gas turbine the exhaust heat can be used to dry it ready for combustion. Two experiments in the use of peat have been undertaken in Britain. Ruston and Hornsby have adapted one of their 750 kW sets to run on this fuel. The wet peat, with 83 per cent moisture content, is passed through an exhaust-heated dryer and then to a cyclone-type separator. At a rate of about 0 5 lb/sec the dried peat, moisture content less than 30 per cent, is fed to the combustion chamber. When practical operation has been proved the steam generated in the process of drying the peat will be utilized to augment the power output.

The other project, by John Brown and Co. Ltd., is for the external combustion of peat. An air boiler fired by peat has been constructed for their 500 h.p. closed-cycle turbine set.

MOBILE GENERATING SETS

For a given output a simple-cycle gas turbine is of relatively light weight and small bulk. As, furthermore, it needs no water supply and can be quickly started and brought up to load, it is admirably suitable as a power unit for a mobile generating set for emergency service. Svenska Turbinfabriks Aktienbolaget Ljungström (STAL) in Sweden build 2,400 kW sets for that purpose, having a five-stage turbine driving a 14-stage axial compressor and the electric generator. Specific fuel consumption is at the rate of 17,060 B.Th.U/kW/hr, representing an overall efficiency of 20 per cent at full load.

Installed in a rail car, of standard 4 ft 8½ in gauge and overall length 67 ft, it forms a self-contained unit which if accompanied by one or more fuel tank wagons is self-sustaining. It can be rapidly moved to areas suffering from disaster or breakdown of usual power supplies and can also be used to take over generation, either wholly or partially, while a static plant is shut down for a periodic overhaul.

Such plants have an obvious military and defence value, particularly if they can be broken down for transport overseas. The U.S.A. is interested in 4,000 kW mobile sets for these purposes.

The Ruston gas turbine, described earlier, is admirably suited for such service. One set for the Air Ministry is specially adapted for transport by air. It is constructed in three independent units, each mounted on its own frame and furnished with bogies for transport from the aircraft to the site. The first includes the compressor, h-p turbine and combustion chamber ; the second has the l-p turbine and the reduction gear ; and the third comprises the alternator and exciters. Assembly is effected by bolting together the three sections, which are levelled and supported on

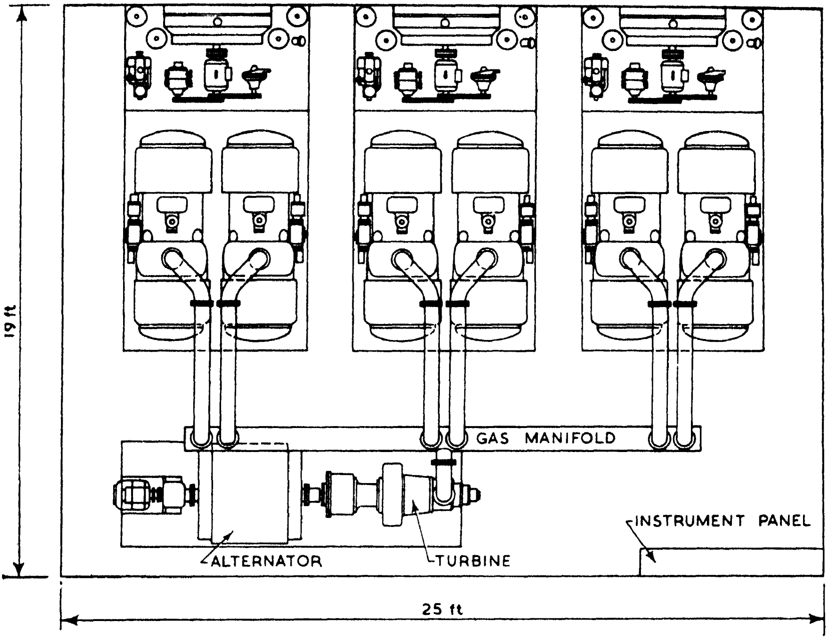
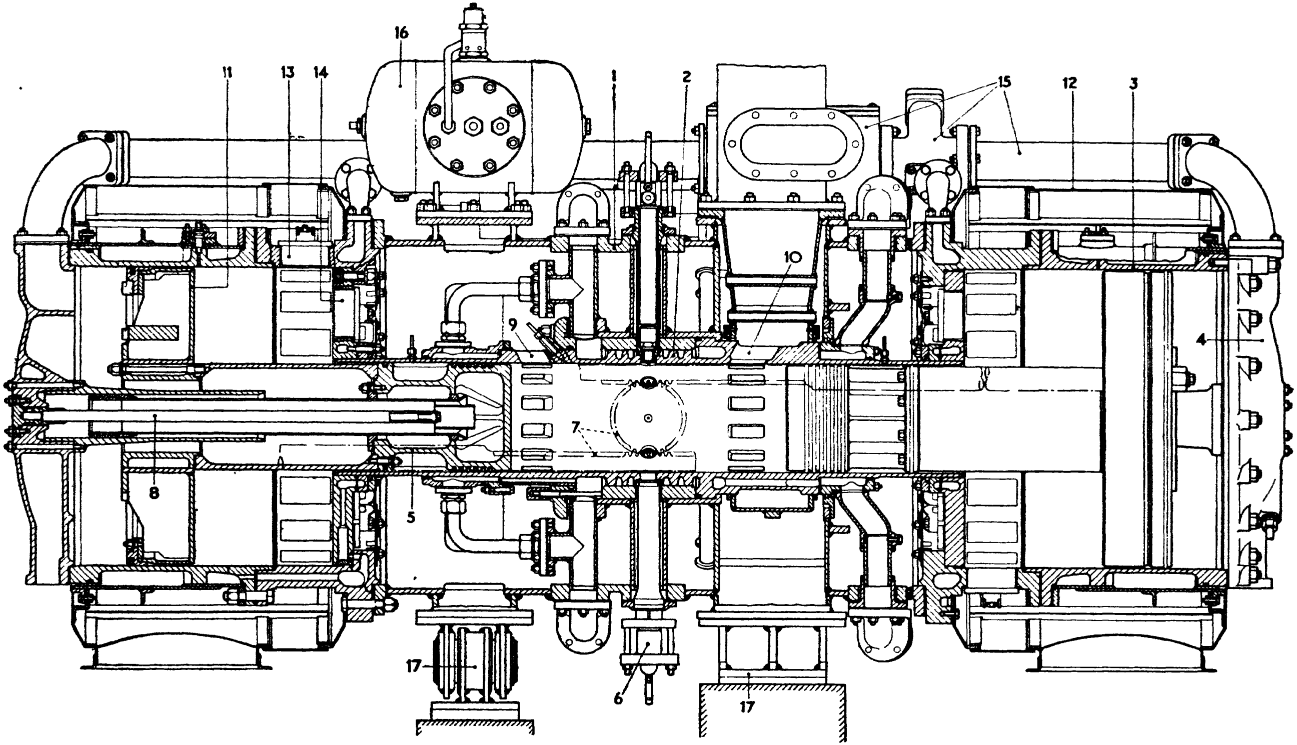


Fig. 17 : 19—Layout of Alan Muntz air-transportable 1,200–1,500 kW electric generating set.

seven screw jacks, and attaching intake and exhaust ducting. A flexible coupling permitting slight misalignment between the work turbine and the alternator is provided. Given a prepared site the set can be assembled, connected up, checked and run in about 24 hr.

Alan Muntz has also developed a free-piston gasifier set for air transportation. The complete unit weighs about 25 tons and is divisible into five component sections, none of which exceeds 5 tons. It can be speedily set up on any roughly levelled site. The plant comprises three twin CS-75 gasifiers and their auxiliaries, each mounted on a bed-plate and weighing 5 tons. A fourth load consists of the turbine, reduction gear, the alternator rotor, and the necessary pipework, while the alternator stator constitutes the fifth load. Output can be arranged from 1,200 to 1,500 kW.

A transportable set may be called upon to run continuously, 24 hr per day, and in an emergency area or at an advanced military base the question of fuel supplies may be of considerable importance. In this respect the free-piston gasifier shows to some advantage by reason of its relatively low rate of consumption and its ability to use fuels both lighter and heavier than the usual diesel fuel. The specific consumption of the Alan Muntz set, on a 24 hr operational rating, is 0·416 lb/s.h.p./hr at the alternator coupling.



1. Engine casing: 2 Engine cylinder: 3. Compressor cylinder: 4. Cushion head: 5. Engine piston: 6. Fuel injector: 7. Synchronizing racks and pinion: 8. Piston cooling oil feed: 9. Inlet ports: 10. Exhaust ports: 11. Compressor and cushion piston: 12. Air intake casing: 13. Air suction valves: 14. Air delivery valves: 15. Cushion control: 16. Starting valve unit: 17. 3-point mounting.

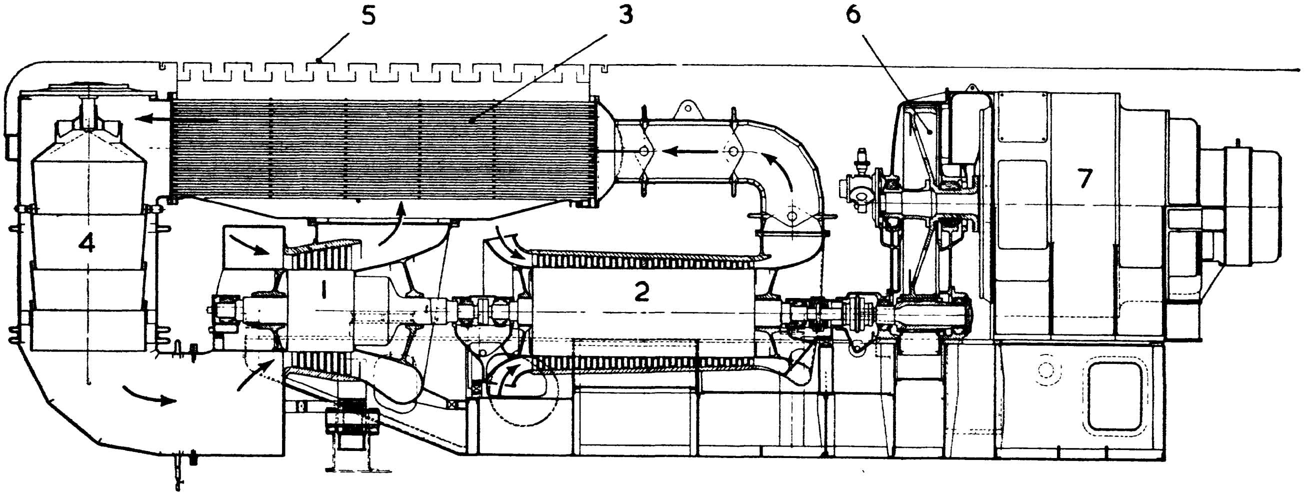
Fig. 17 : 20—Alan Muntz CS-75 free-piston gas generator.

FREE-PISTON GASIFIERS

Although involving a reversion to reciprocating motion, the free-piston charging set enables high thermal efficiencies to be realized. Within the limits imposed by available materials, the fuel can be burned at a higher pressure and the operating gas temperature can be higher in an intermittent cycle than in the continuous combustion cycle of the pure gas turbine. The free-piston air compressor—that is a single unit in which opposed pistons in a diesel two-stroke cylinder carry directly connected, co-axial compressor pistons—has long been used in Europe. No crank or connecting rod is used, the pistons being returned by air compressed in ‘ cushion ’ or ‘ return ’ cylinders and only coupled together by a light linkage or racks and a pinion to synchronize the motion. Length of stroke and rapidity of reciprocation is determined by regulation of the cushion cylinder mean pressure. Employed as a gasifier for a turbine, all the air compressed is used to scavenge and charge the engine cylinder. Only about 40 per cent of the air is burned and the remainder serves to dilute and cool the combustion gases which are ducted to an exhaust gas turbine producing the useful output. The gasifier unit operates at an efficiency of from 40 to 45 per cent and thus, in conjunction with a turbine of present-day efficiency, an overall thermal efficiency of about 35 per cent can be achieved.

During the period of the Second World War the Pescara organization in France had a 750 kW set running at Belfort, Alan Muntz & Co. Ltd. of Heston was developing, under a Pescara licence, a 400 gas h.p. unit for the Admiralty, and in Switzerland Sulzer Bros. built a 7,000 h.p. set with three gasifiers feeding a single turbine. In the U.S.A. both the Lima Hamilton Corporation and the Cooper-Bessemer Corporation after the war undertook the development of free-piston gasifiers for marine and rail applications. By agreement with Alan Muntz and the French interests, certain rights in the Muntz-Pescara patents and designs have been acquired by the General Motors Corporation. As the world's largest builders of diesel-electric locomotives the interest is presumably directed towards the locomotive application. Units of the type shown in Fig 17 : 20 and built by Alan Muntz have been shipped to America for exhaustive test and analysis. These have engine and compressor cylinders of 7·5 in and 20·75 in respectively and develop the rated output of 450 gas h.p. with a stroke of 10 in and reciprocation at about 1,100 cycles per min.

In France development has been most actively promoted by the Société d'Etudes Mécaniques et Energetiques (S.E.M.E.) and units are now being produced in series by the Société Industrielle Générale de Mécanique Appliquée (S.I.G.M.A.) for stationary, rail and marine installations. A standardized unit, the GS.34, with an engine cylinder 13·38 in diameter, compression cylinders 35·43 in diameter, and an effective



1. Turbine: 2. Compressor: 3. Heat exchanger: 4. Combustion Chamber: 5. Exhaust: 6. Reduction gear: 7. Generator.

Fig. 17 : 21—Layout of Brown-Boveri 2,200 h.p. set for the world's first gas turbine locomotive.

stroke of 17·7 in, has a mass flow of 8·1 lb/sec at 585 reciprocations per minute. The gas is delivered at about 58 lb/in^2 abs pressure and 480° C temperature and the output of the turbine is 1,230 b.h.p. Fuel consumption on test was 0·332 lb/h.p./hr and in service generating electricity the overall consumption is about 0·37 lb/h.p./hr.

GAS TURBINE LOCOMOTIVES

Despite the growing competition of road and air services the railways remain indispensable for the overland transport of passengers, food, merchandise and raw materials to the extent found necessary in a modern industrialized country. As such a heavy investment is required in tracks, depots, signal and safety systems, and rolling stock, managements must continuously endeavour to reduce working and maintenance costs in order to remain competitive with other means of transport. The ubiquitous steam locomotive has been developed to a high standard, electric traction has proved efficient in services carrying a traffic sufficiently dense to justify the expenditure on power lines, and the diesel-electric loco has established itself for high-speed long-distance service on non-electrified tracks. Naturally, the advent of the reliable gas turbine has not been neglected by railway engineers.

Potentially, the gas turbine locomotive would appear to offer many advantages. According to reliable American estimates it should be regarded as superior to the steam locomotive in all respects save that of first cost, which it exceeds, and of total working life, which it equals. This contrasts with the diesel-electric locomotive which is held to be inferior to the steam locomotive as regards first cost, working life, maintenance, and lubrication costs. In spite of these aspects the diesel-electric locomotive is in the U.S.A. being built at the rate of about three to one steam locomotive.

Comparing the gas turbine with the diesel-electric unit presents a different but not unsatisfactory picture. It shows to advantage in all respects except drawbar efficiency and time for refuelling. Such comparisons, of course, are necessarily of a generalized character and may be modified by specific economic or operational conditions. They assume, for instance, production in series and operation on bunker oil costing only 50–60 per cent of diesel oil. Nevertheless, it must be remembered that the gas turbine is at the beginning of its development and will undoubtedly be materially improved in the future. The maximum gas temperature at the turbine nozzle was conservatively taken at about 600° C yet aviation turbines are already operating at over 800° C with 600–1,000 hr between overhauls.

Absence of reciprocating parts implies less maintenance, greater availability for duty and longer life. Further, the unit operates well below its critical speed and the locomotive is thus suitable for high-speed operation. A 2,500 h.p. gas turbine locomotive may be 20 tons lighter than an equivalent diesel-electric unit and consequently track wear and wheel wear may be expected to be lessened. No water is required, thus obviating the frost hazard, facilitating operation in arid climates and reducing or eliminating maintenance of water installations. The gas turbine is quickly available for duty as no time is required for raising steam and no extensive warming-up period is necessary.

It was in the autumn of 1941 that the trials of the first gas turbine locomotive were run. This was the Brown-Boveri 2,200 h.p. unit for the Swiss Federal Railways. The power plant was a simple, open-cycle set with a five-stage turbine driving an 18-stage compressor which takes in air at the rate of 62 lb/sec (48,590 ft^3/min), raises the pressure to about 57 lb/in^2 abs, and delivers it through the heat exchanger to the single vertically mounted combustion chamber. Gas temperature at the nozzle is limited to about 600° C and the turbine efflux passes through the heat exchanger and is exhausted through the roof. At 5,200 r.p.m., the maximum operational speed, the turbine develops 10,000 h.p. of which 7,800 h.p. is absorbed in driving the compressor. Useful output of 2,200 h.p. is taken from the end of the compressor shaft to drive, through a 6·4 : 1 reduction gear, the main D.C. generator and two smaller generators for auxiliary services.

Assembled on a fabricated base frame, the complete power plant is mounted in the locomotive with a three-point elastic suspension. The locomotive drive is by four traction motors at the driving axles. Apart

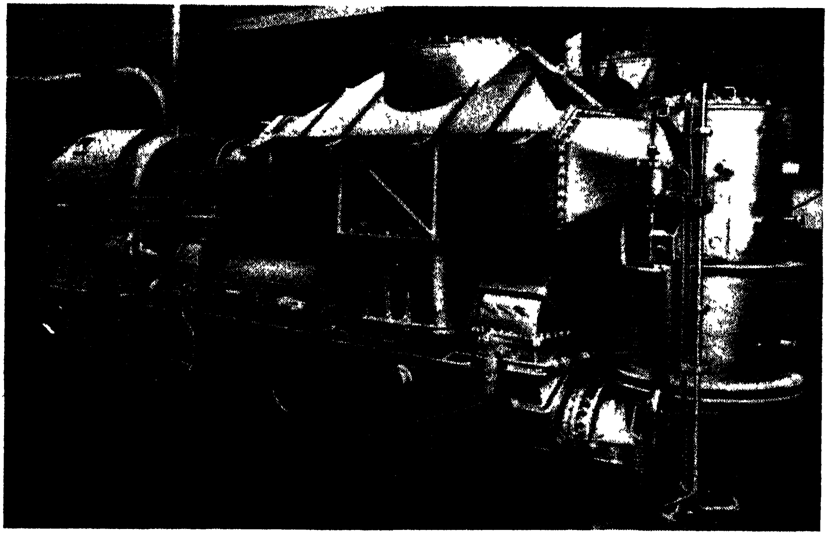


Fig. 17 : 22—Brown-Boveri 2,200 h.p. locomotive set on test bed.

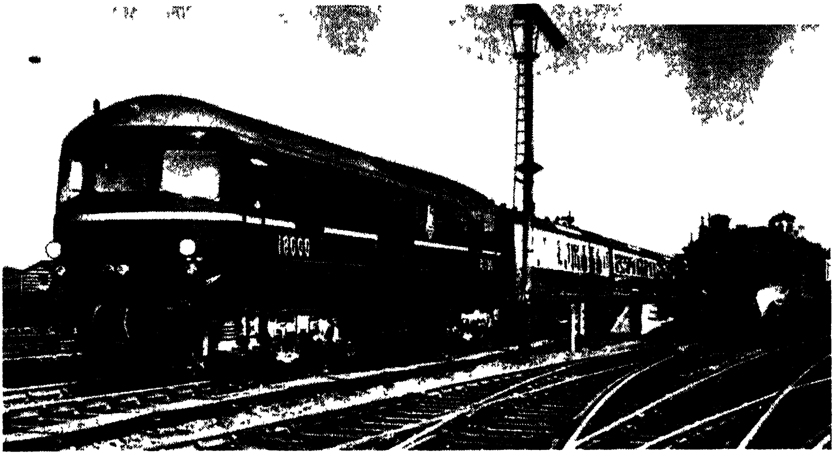


Fig. 17 . 23 -In scheduled service on the British Railways (Western Region), the 2,500 h.p Brown-Boveri gas turbine locomotive.

from a period during the Second World War when a supply of oil fuel was not available the locomotive has been operated regularly on a main-line passenger service by the Swiss Federal Railway.

Late in 1946 the former Great Western Railway decided to build two gas turbine-electric locomotives for main-line passenger service. They were intended for runs from London to Plymouth, Bristol, Fishguard and Birkenhead on which trains to a maximum of 18 coaches weighing 650 tons would be required to attain speeds up to 90 m.p.h. One was ordered from British Brown-Boveri Ltd., and the other from Metropolitan Vickers Electrical Co. Ltd.

Built in Switzerland, with the chassis and running gear by the S.L M. Works, the Brown-Boveri locomotive ran its trials in the summer of 1949 and was delivered to the British Railways (Western Region) later that year. The power plant closely followed that of the earlier Swiss unit but the useful output was raised to 2,500 h.p. As before, a four-axle drive was provided but the wheel arrangement was completely different. The British scheme with two three-axle bogies each driving on the outer axles (A1A—A1A) was specified as being more suitable for high-speed operation than the Swiss arrangement (1A—B_0—A1).

The complete locomotive weighs 115 tons, has a tractive effort of 32,600 lb from starting up to 21 m.p.h. and a continuous tractive rating of 12,600 lb at 64 m.p.h. Controls are duplicated for driving cabs at each end to permit running in either direction without turning. A small diesel-engined generator set, provided for starting and auxiliary services, can be used for manœuvring the locomotive without starting up the gas turbine. Steam for train heating is raised in a small, oil-fired boiler. To

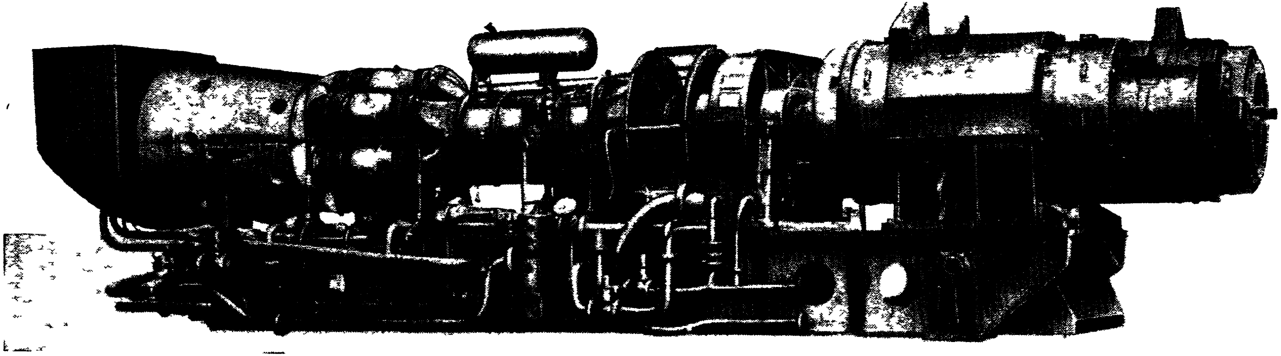


Fig 17 24—Gas turbine-generator set for Metrovick locomotive

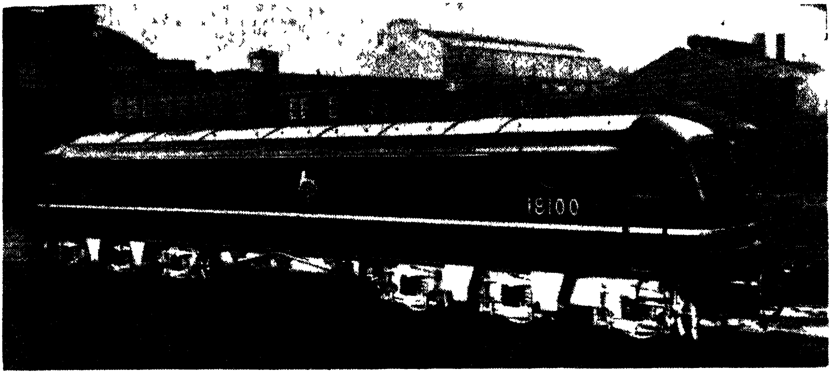


Fig. 17 : 25 —Metrovick 3,000 h.p. gas turbine-electric locomotive for British Railways (Western Region).

suit British rolling stock equipped with vacuum braking, two exhausters driven respectively by an electric motor and by vee-belts from the main generator shaft are fitted instead of the compressor used for the Continental pressure braking system.

The fuel used is a heavy boiler oil having a calorific value of 18,760 B.Th.U./lb; specific gravity is 0·945 and viscosity 910 sec Redwood. Steam heating is necessary at the tanking stations in cold weather to ensure the fuel is sufficiently fluid to flow through the pipelines.

For the second locomotive the Metropolitan-Vickers Co. built a power plant of different type designed for maximum output of 3,500 h.p Drawing upon its experience in the development of the F.2 aircraft turbojet, a satisfactory thermal efficiency is attained without the aid of a heat exchanger by employing a relatively high compression ratio and a high gas temperature at the turbine nozzle.

The ‘straight-through’ turbine set runs at 7,000 r.p.m., a 15-stage compressor driven by a five-stage turbine handling an air mass flow of 50 lb/sec. Compression ratio is 5 25 : 1 and the air is delivered to six short tubular combustion chambers grouped around and parallel to the main axis. This combustion arrangement becomes possible as the fuel used in this instance is a relatively light gas oil.

Actually the maximum output is higher than is required for the duty and the loco set is accordingly arranged for a maximum of 3,200 h.p. and a continuous rating of 3,000 h.p. Reduction gear and auxiliary equipment absorb about 150 h.p. leaving for the generators about 2,850 h.p. A pinion mounted on an extension of the compressor shaft drives on each side a shaft running at the reduced speed of 1,600 r.p.m. One of these drives two of the main generators in tandem while the other drives the third main generator, the auxiliary generator and an exciter for the main units.

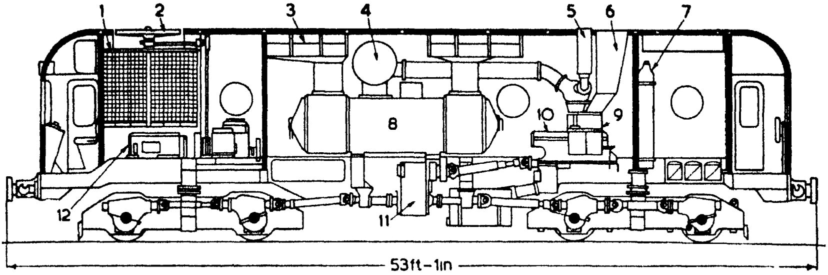
For this locomotive the wheel arrangement is also twin three-axle bogies but all six axles are driven (C_0—C_0). Each main generator feeds two traction motors and the final gearing is determined to give a maximum tractive effort of 60,000 lb and a continuous rating of 30,000 lb. Length over buffers is 66 ft 9 in and weight in running order is 129·5 tons. Starting is effected by using the main generators as motors fed from the 384 amp/hr storage battery.

In the U.S.A. the General Electric Co. in conjunction with the American Locomotive Co. built a 4,800 h.p. gas turbo-electric locomotive for freight haulage. The prototype, first run in November 1948, was operated experimentally on several different railroads and in two years had logged more than 100,000 miles of service running, mostly on heavy oil fuel. Despite some initial troubles and minor failures mainly confined to combustion chambers and blades, the performance was satisfactory and the Union Pacific Railroad ordered ten locomotives. The first of these production units was delivered early in 1952, the order has now been completed and a further 15 have been ordered.

The power plant is of the simplest open-cycle type without heat recuperation. A 15-stage compressor handles an air mass flow of 94 lb/sec at a pressure ratio of 5·9 : 1 and delivers to six tubular combustion chambers. Gas temperature at the nozzle of the two-stage turbine is 705° C at the usual rating but can safely be raised up to a maximum of 760° C if required. Running at 6,700 r.p.m. the turbine drives the four generators through a reduction gear at 1,600 r.p.m. The loco has four two-axle bogies and drives on all axles (B_0—B_0—B_0—B_0). Control cabs were provided at each end of the prototype but this feature was found to be unnecessary for freight service and the production version has only a single cab.

Westinghouse Electric Corporation has also built a 4,000 h.p. locomotive of the four-bogie type driving on all axles. In this two gas turbo-generating sets of 2,000 h.p. each are installed side by side. They are of open-cycle, single-shaft type without heat exchanger and run at 8,750 r.p.m. A 23-stage compressor delivers air at a pressure ratio of 5 : 1 to 12 combustion chambers burning heavy oil fuel. An eight-stage turbine drives tandem generators at a gear-reduced speed of 1,150 r.p.m. The air mass flow is 32 lb/sec and the maximum gas temperature is 732° C. The locomotive has been in trial service on several railroads.

An interesting small locomotive of unusual type has been developed by the Renault concern in France. A six-stage Rateau turbine is fed by a S.I.G.M.A. (Société Industrielle Générale de Mécanique Appliquée) free-piston gasifier of the standard GS.34 type described earlier in this chapter. The gasifier operates at about 600 reciprocations per min while the turbine, producing 1,200 h.p. at 12,300 r.p.m., is rated to deliver 1,000 h.p. at 8,500 r.p.m. The gas temperature at the turbine entry is



1. Radiators: 2. Fan: 3. Air filters: 4. Gas chamber: 5. Turbine by-pass: 6. Turbine exhaust: 7. Compressed air cylinders: 8. Gasifier: 9. Turbine: 10. 6 : 1 reduction gear: 11. 2-speed gearbox: 12. Diesel auxiliary engine.

Fig. 17 : 26—Renault 1000 h.p. free-piston gasifier locomotive.

only 475° C at the rated output. From the turbine the transmission is through a 6 : 1 reduction gear to a central chassis-mounted two-speed gearbox and thence fore and aft by propeller shafts to the two four-wheel bogies. All axles are driven.

A tremendous amount of study and experiment has been directed in both Britain and the U.S.A. to the problem of running gas turbine locomotives on solid fuel. Coal-burning is almost an essential in Britain and desirable in certain regions in America. To date no coal-fired locomotive has gone into service but most promising developments have been operated on the test bed.

Probably the most sustained effort has been on the two experimental turbo-electric locomotives with open-cycle, single-shaft, recuperative sets ordered by the Locomotive Development Committee of Bituminous Coal Research, Inc. Both have been delayed while development of the coal-firing equipment proceeds. Turbines and generators have been built for some considerable time. The Elliott plant, for a loco to be built by the Baldwin Locomotive Works, comprises a two-stage centrifugal compressor with a pressure ratio of 3·5 : 1, heat exchanger, single combustion chamber and four-stage turbine with gas entry at 690° C. It is rated at 3,910 h.p. at 5,910 r.p.m., the single generator being driven at 900 r.p.m.

Of the same basic type, the Allis-Chalmers set for a loco by the American Locomotive Co. develops 4,250 h.p. at 5,700 r.p.m. The 21-stage axial compressor has a ratio of 4·8 : 1 and feeds air at the rate of 66 lb/sec through the heat exchanger to twin combustion chambers equipped with fly-ash separators. Maximum gas temperature at the nozzle of the six-stage turbine is 705° C. Four generators are driven through reduction gearing at 2,000 r.p.m.

The locomotive is of the two-car type, the main power plant in one car and the coal bunker, pulverizing mill, train-heating boiler and auxiliaries in the other, with a control cab at the extremities. Each car is on twin three-axle bogies and a total of eight axles are driven, the arrangement being 2(A1A—A1A).

After a prolonged investigation of coal-burning problems by Pametrada (Parsons and Marine Engineering Turbine Research and Development Association), the Ministry of Fuel and Power has sponsored the building of a novel locomotive for experimental operation. The open-cycle, two-shaft, turbine set will be of the external-combustion type and thus any fouling, corrosion or erosion effects arising from the combustion of coal will be confined to the ' air boiler ', while the turbines will operate on clean hot air.

An axial compressor delivers air under pressure to a nested-tube heater from which it is ducted at a maximum temperature of 705° C to the h-p turbine driving the compressor. After further expansion in the independent l-p turbine providing the useful power output, the still-heated air is fed to the coal-burning chamber. The combustion gases are cleaned, passed through the air heater and finally exhausted by way of a train-heating boiler.

Mechanical transmission has been selected for this locomotive. The output turbine, running up to 8,000 r.p.m. at the rated maximum of 1,800 h.p., drives through a reducing, two-speed gearbox and fore and aft propeller shafts to six-wheel coupled bogies. Power plant will be built by C. A. Parsons and Co. Ltd. and the chassis and running gear by the North British Locomotive Co. Ltd.

MARINE PROPULSION

The problems involved in the application of the gas turbine to large passenger or cargo vessels are being intensively and continuously investigated. Shipowners are rightly conservative, and reliability, consistent economic operation, continuous availability between periodic overhauls, and long working life must be positively assured before gas turbines are adopted for this vital service.

Outstanding enterprise has been shown by the Shell Petroleum Co. Ltd. The Anglo-Saxon 12,290 tons tanker *Auris*, specially built for the purpose and powered by four diesel engines generating power for the electric drive, was used as a test vehicle for a marine gas turbine. A gas turbo-alternator to replace one of the diesel units was built to specific requirements by the British Thomson-Houston Co.

Of the open-cycle type with heat exchanger the set may be classed as of ' heavy ' construction, as distinct from the ' light ' sets developed for naval duties. The order was placed in 1946 and the set was first run on test in 1950. Initial operation was on gas oil but later a heavy boiler oil was used. An air mass flow of 28 lb/sec is raised to a pressure of

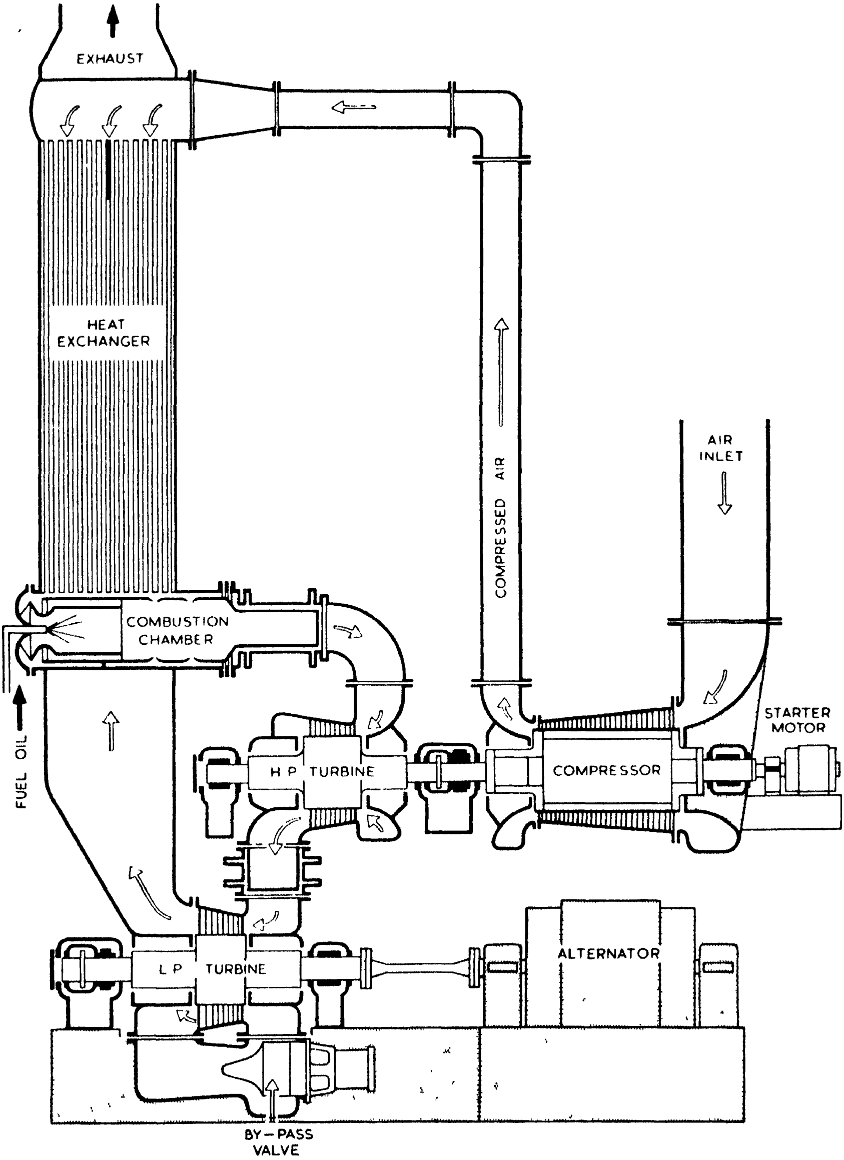


Fig. 17 : 27—Diagrammatic layout of Auris *plant.*

about 60 lb/in² abs by a 24-stage compressor and passed through the tubes of a vertical, contra-flow heat exchanger to two horizontal combustion chambers. The combustion gases have a temperature of about 600° C at the nozzle of the seven-stage h-p turbine which drives the compressor. Leaving the h-p unit at about 447° C they pass through the six-stage l-p turbine driving the alternator and discharge around the tubes of the heat exchanger to the exhaust.

On test in January 1951 on heavy boiler oil—calorific value 17,650 B.Th.U./lb and viscosity 1,500 sec Redwood—the output was 1,022 kW at the alternator terminals for a fuel consumption of 0·905 lb/kW/hr, corresponding to an overall thermal efficiency of 21·4 per cent. The h-p set runs at 6,000 r.p.m. and the l-p turbine and alternator at 2,940 r.p.m.

The installation in the tanker—in substitution for one of the diesel sets—was made later in the year and immediately after running 48 hr sea trials the vessel sailed for Tampico on normal operation. Early in the following year, 1952, an experimental westward run from Plymouth to Curaçao was made solely on gas turbine power. Despite several days of heavy weather the turbine ran continuously without varying speed for 543 hours. For the 3,897-mile voyage the ship's speed averaged 7·35 knots. (With all four power units operating the ship has a speed of 12 knots.) The fuel used throughout the voyage was a boiler oil of specific gravity 0·958 and viscosity 1,300 sec Redwood.

Following this experience the Shell organization decided to build a tanker powered solely by gas turbines. The operating company, the Anglo-Saxon Petroleum Co. Ltd., early in 1953 placed an order with Cammell Laird of Birkenhead for a single-screw vessel of 18,000 tons deadweight to be completed in 1956. Turbine units will be built by the British Thomson-Houston Co.

Two gas turbo-alternators will drive the propeller through two electric motors and a two-pinion, single reduction gear. Power for all auxiliary requirements at sea and in port, including that for the cargo pumps, will be drawn from the propulsion units. These are of the open-cycle, compounded type with intercooling and heat recuperation. The h-p line will comprise turbine, compressor and geared auxiliary alternator with exciter while in the l-p line is a turbine, compressor and propulsion alternator.

A diesel alternator set will furnish power for ship services when the gas turbines are not in operation. Steam for cargo heating and tank cleaning will be raised in a small oil-fired boiler and in waste heat boilers taking exhaust from the turbine heat exchangers. Steam turbines, each of 150 h.p., will be used for initial starting of the main gas turbine sets.

In both Britain and the U.S.A. other ' heavy ' marine propulsion sets have been built and tested. They are being used for experiment, study,

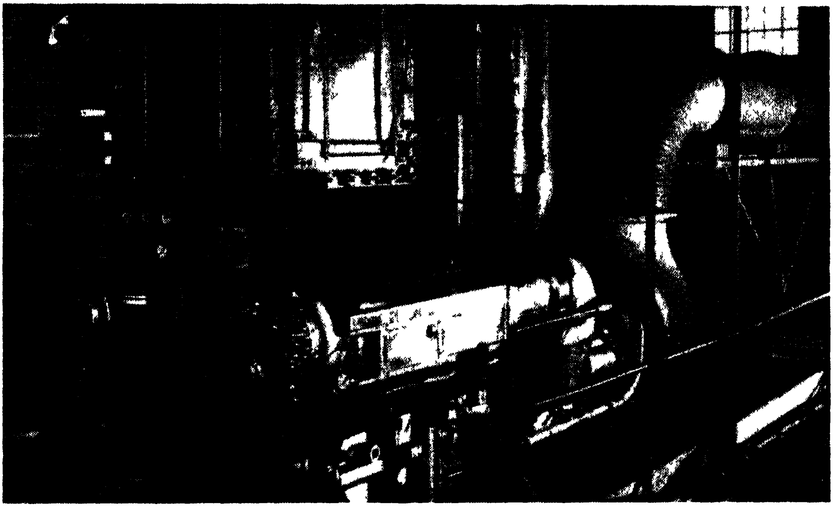


Fig. 17 : 28 – 1,000 kW set for Anglo-Saxon tanker Auris on test at the B.T.H. works. View shows, right to left, compressor, h-p turbine, combustion chambers and base of heat exchanger. The l-p turbine and alternator are immediately below.

and to gain operating experience and are not likely to be installed in a ship. The largest of these is the English Electric 6,500 h.p. unit built as a development project for the Admiralty and first run in September 1951. A parallel-flow layout was adopted, the delivery of a single compressor being divided between two combustion chambers feeding independent charging and output turbines. This arrangement makes possible a small, high-speed output turbine suitable for direct connection to an existing standard marine alternator and also facilitates manœuvring as control can be effected simply on the output set while leaving the charging set undisturbed.

An air mass flow of 128 lb/sec is delivered by the 15-stage compressor at a pressure ratio of 4 : 1. The major portion, 86 lb/sec, passes through a heat exchanger to four combustion chambers for the five-stage turbine driving the compressor at 4,000 r.p.m. The remainder flows through another heat exchanger and two combustion chambers to the five-stage output turbine which drives the alternator at 5,600 r.p.m.

A commendable standardization of component units was effected in this design. The two turbines are geometrically similar, the output turbine being scaled down in a linear ratio of 0·688 : 1. Combustion chambers are identical. Tube stacks for the heat exchangers, each comprising 1,800 tubes about 11 ft long, are interchangeable. Eighteen stacks are required, 12 for the charging set and six for the power set.

The peripheries of the turbine rotors are shielded and both the peripheries and the blade roots are cooled by air tapped from the compressor. This abstracts somewhat less than 2 per cent of the cycle air flow, but, it is claimed, maintains the rotors at a temperature 250° C lower than the gas temperature, 704° C, at the nozzles.

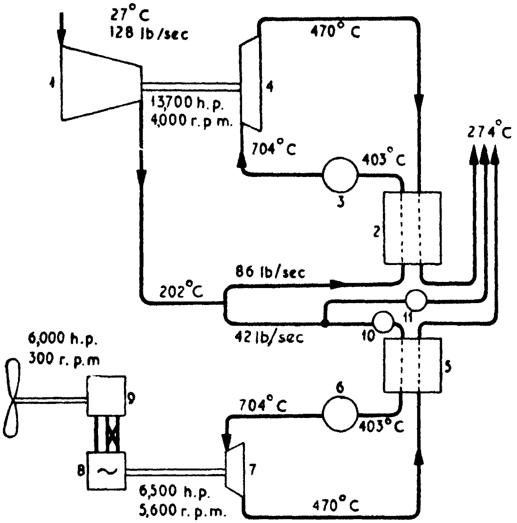
A 3,500 h.p. set built by Allis-Chalmers was installed at the U.S. Naval Engineering Experiment Station, Annapolis, in 1944. The objective was to develop a 'long-life' marine plant that could be operated at the high gas temperatures used on the relatively 'short-life' aircraft units. Designed for a maximum gas temperature of 815° C, the early 50 hr tests in 1945 were run successfully with temperatures of 650° C and 735° C. These were made on No. 2 furnace oil, a distillate, but the intention was to use a heavy fuel oil after the proving tests.

To meet naval requirements of rapid manœuvre, the cycle selected was of the parallel-flow type. A 20-stage compressor with a pressure ratio of 4 : 1 passes air round the tubes of the heat exchanger and thereafter the flow is divided. About 31·5 lb/sec is delivered to one combustion chamber for the turbine driving the compressor and 19·0 lb/sec to the other chamber for the power turbine. The effluent of both turbines is discharged through the tubes of the heat exchanger.

Both turbines run up to a maximum speed of 5,200 r.p.m. They are similar units, each with two shrouded impulse stages and three reaction stages. A novel method of cooling the first-stage wheel is provided. Co-axial tubes bring a supply of air to the upstream face of the disc. It is discharged through jets on to the rim, flows rapidly inwards and then passes out through the bore of the centre tube to a water-cooled exchanger and is recirculated by a blower. Another supply of air, fed to the hollow diaphragm carrying the second-stage nozzle ring, discharges on to the downstream face of the disc and is expended through the diaphragm labyrinth seal.

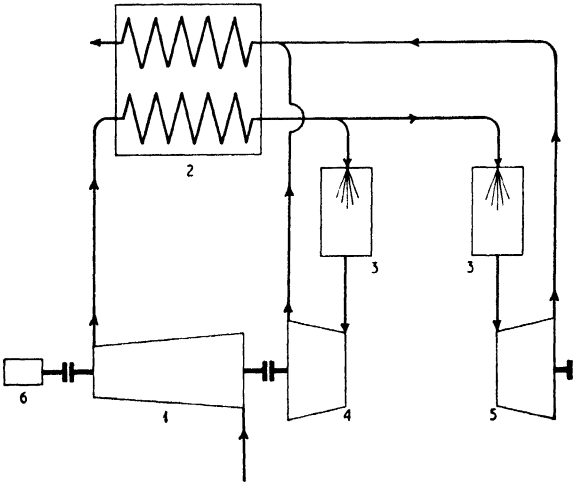
Another pioneer American experimental plant was the 2,500 h.p. set built by the Elliott Co. and first run on the test bed in 1944. A compounded open-cycle arrangement was selected with intercooling, reheat and recuperation. An unusual feature was the use of positive-displacement compressors of the Lysholm twin-rotor helical-lobe type. In these units meshing clearance between pairs of rotors is maintained by gears, so the working surfaces require no lubrication. The air mass flow is about 30 lb/sec and the l-p compressor, a duplex unit with two pairs of rotors, is rated to deliver 25,000 ft^3/min at a pressure ratio of 3 : 1 at about 4,000 r.p.m.

From the l-p compressor the air passes through the intercooler to the h-p compressor, and thence through the tubes of the heat exchanger to the combustion chamber feeding the h-p turbine. On leaving the h-p turbine, which drives the l-p compressor, the gases are reheated in a second combustion chamber before delivery to the l-p turbine driving



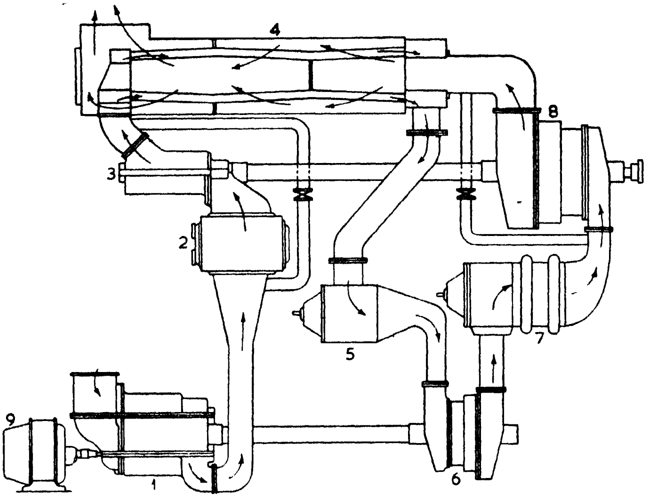
1. Compressor: 2. Heat exchanger: 3. Combustion chambers: 4. Charging turbine: 5. Heat exchanger: 6. Combustion chambers: 7. Power turbine: 8. Alternator: 9. Propulsion motor: 10. Throttle valve: 11. Blow-off valve.

Fig. 17 : 29—Diagram of parallel-flow layout of English Electric marine set.



1. Compressor: 2. Heat exchanger: 3. Combustion chamber: 4. Charging turbine: 5. Power turbine: 6. Starting motor.

Fig. 17 : 30—Cycle diagram of 3,500 h.p. Allis-Chalmers set for the U.S. Navy.



1. L-p compressor: 2. Intercooler: 3. H-p compressor: 4. Heat exchanger: 5. Combustion chamber: 6. H-p turbine: 7. Reheat chamber: 8. L-p turbine: 9. Starting motor.

Fig. 17 : 31—Schematic layout of 2,500 h.p. Elliott-Lysholm marine set.

the h-p compressor at about 3,000 r.p.m. and providing the useful power output. The effluent is discharged through the heat exchanger to the exhaust stack. On test, efficiencies of 29 per cent and 20 per cent at full load and quarter-load respectively were reported.

LIGHTWEIGHT MARINE UNITS

Relatively light marine propulsion sets, largely based on designs proved in aircraft units, are being developed and installed experimentally in naval vessels. Actually, a British Navy gunboat, M.G.B. 2009, with a Metrovick set of this type was the first gas turbine-powered vessel to be taken to sea. Successful sea trials were run in July 1947 in the Solent, where 50 years earlier the *Turbinia*, the first steam turbine-driven vessel, did her trials.

Designated the ‘ Gatric ’, the 2,500 h.p. plant comprised a gas generating set—ten-stage axial compressor, annular combustion chamber, and two-stage turbine drawn from units under construction for aircraft—and in series an independent four-stage output turbine driving the propeller through a reduction gear.

Although purely an experimental installation the results were impressive. The gas turbine replaced a standard petrol engine of 1,300 s.h.p. and thus demonstrated the possibility of turbine machinery giving high power output while requiring reduced space. It showed to advan-

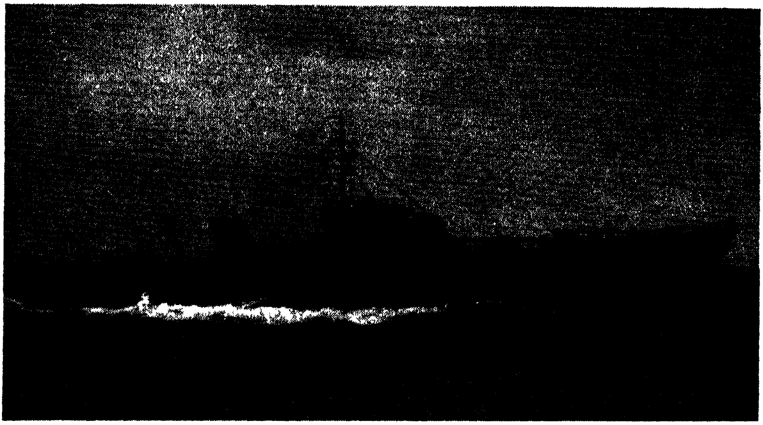


Fig 17 32—The first vessel in the world to be powered by a gas turbine The M G B 2009 with Metrovick Gatric power unit

tage as regards weight, even when compared with the petrol engine The specific weights per shaft horse-power for the unit alone and with reduction gear were 1 74 lb and 2 78 lb while for comparable output with two petrol engines the figures were 2 25 lb and 3 01 lb

As a result of the successful operation of the Gatric unit at sea the Royal Navy ordered four larger sets from the same manufacturer The U S N also ordered two sets for test and evaluation Designed for a maximum output of 4,800 s h p the ‘ G2 ’ sets will be operationally rated at 4,500 s h p maximum and 4,000 s h p continuous Of the same ‘ straight-through ’, open-cycle type without heat exchanger as the

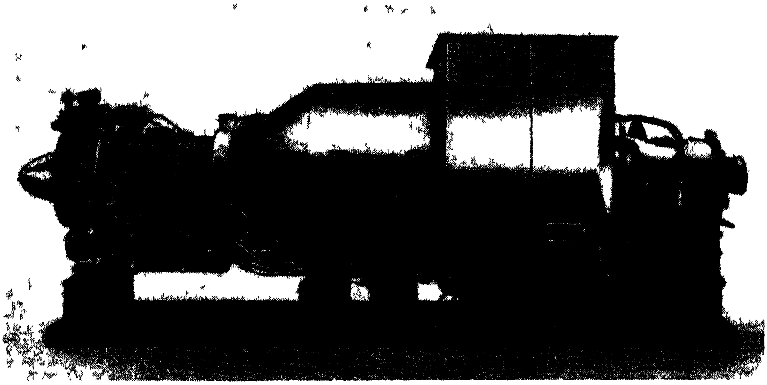


Fig 17 : 33—Metrovick 4,500 s.h.p. G2 marine propulsion set for naval gunboats

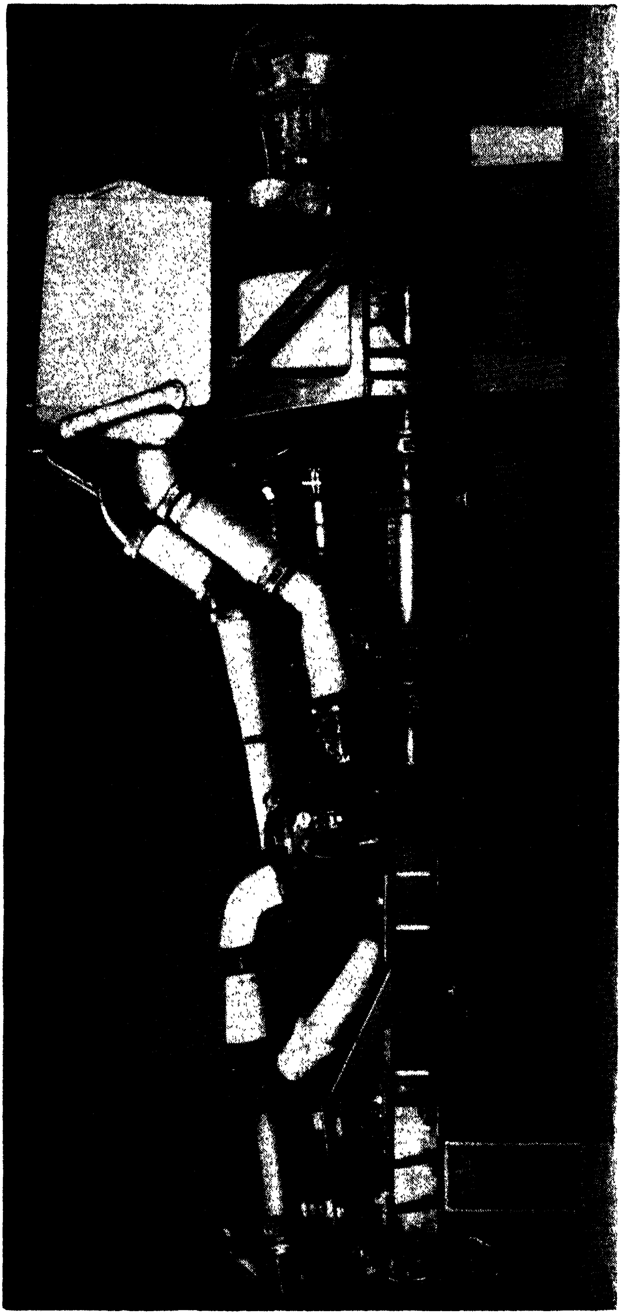


Fig. 17 : 34—Rolls-Royce 6,000 s.h.p. compounded-compression lightweight marine set.

Gatric, the G2 units have a reduced specific weight, the figures being 1·54 lb and 2·16 lb for the unit only and complete with reduction gear.

A slightly modified version of the Beryl turbojet aircraft unit gas generating set runs at a maximum speed of 7,830 r.p.m. The 11-stage compressor feeds an air mass flow of 62·6 lb/sec at a pressure ratio of 4·03 : 1 to an annular combustion chamber, of Firth-Vickers Immaculate 5 stainless steel, into which the fuel, Pool gas oil, is injected upstream by 20 burners. Gas temperature at the nozzle of the single-stage turbine is 806° C and Nimonic 80 alloy is used for the guide vanes and Nimonic 80A for the rotor blades. The independent power turbine is a three-stage unit running at 5,200 r.p.m. with an inlet gas temperature of 660° C.

Combustion chamber and turbines are enclosed in a light-alloy casing. The exhaust from the power turbine creates an extractor effect and air is drawn from the engine room through the casing, cooling the turbine cylinders and carrying away fumes. The power turbine drives the propeller at a maximum speed of 1,100 r.p.m. through a single reduction, side-pinion gear.

The G2 units have been installed in Royal Navy motor torpedo boats, two in the *Bold Pioneer* and two in the *Bold Pathfinder*.

Rolls-Royce has also developed marine units for the Royal Navy, the first being run on the test bed in June 1951. Initial installation of two of these sets, designated R.M.60, was in the *Grey Goose*, a 35 kt gunboat originally powered by two steam turbines each of 4,000 h.p. The gas turbines give about 50 per cent increase in power output, a 50 per cent reduction of machinery weight, and also a saving in space occupied. Two units have also been ordered for the U.S. Navy Department.

Of the open-cycle, compounded-compression type the arrangement is unusual in the use of three independent turbines in series. Air is compressed in a l-p axial unit and then, after passing through a sea-water intercooler, in a two-stage centrifugal compressor also furnished with an intercooler. It then flows through a diffusing duct to the heat exchanger and thence to two tubular combustion chambers. The h-p turbine drives the centrifugal compressor, the m-p turbine provides the useful output while the l-p turbine drives the axial compressor. H-p and m-p shafts are co-axial and the l-p shaft runs forward below the heat exchanger. A two-stage gear reduces the power shaft speed for the final drive to a Rotol three-bladed, variable-pitch and reversing propeller. The overall length of the set, as illustrated, is about 30 ft.

For the propulsion of such small craft as launches, pilot boats and harbour boats, small gas turbines are available. Marine versions of the Rover turbine are produced and have been installed and operated experimentally in special-purpose craft. Two were fitted in the 60 ft launch *Torquil*, first run on trial at Anglesey in May 1950. Designed for a

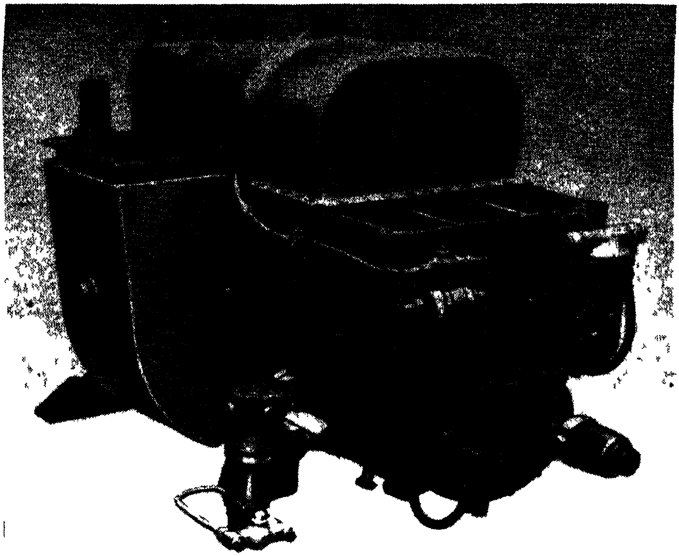


Fig 17 . 35—Rover 150 h.p marine turbine unit.

maximum output of 200 h.p. these units were rated at 150 h p maximum and 100 h p. continuous. Using diesel oil the consumption at full load was 1 25 lb/h.p./hr and the desirability of incorporating a heat exchanger in the cycle was recognized. By that means it was estimated the specific consumption could be reduced to 0 8 lb/h.p./hr.

The charging set comprises a single-stage centrifugal compressor driven at 40,000 r.p.m. by a single-stage turbine and feeding two combustion chambers. In series with the charging turbine is the independent power turbine running at 30,000 r.p.m. and driving the propeller through a 7 : 1 single-helical reduction gear. Starting was effected by a standard car-type electric starter. Each unit weighed approximately 4 cwt, and the overall dimensions were about 2 ft 6 in high, 2 ft 6 in wide and 4 ft long.

In the U.S.A. the small Boeing turbine—the Model 502 two-shaft version—has also been installed in small marine craft. Early in 1951 it was reported that a boat powered by this unit was tested by the U.S.N. at Annapolis at speeds up to 21 kt.

APPENDIX

Early British Developments

Previous editions of this work have dealt at some length with pioneer British developments, and a summary of these is desirable if a more detailed study of current British practice is undertaken. Two lines of development, covering centrifugal compressors and axial compressors, can be distinctly traced. The first is associated with the name of Air Commodore Sir Frank Whittle and the second with Dr. A. A. Griffith.

While a flight cadet at the R.A.F. College in 1928, Whittle discussed in a scientific thesis the possibilities of jet propulsion and of gas turbines, but it was not until 18 months later that he conceived the idea of using a gas turbine for jet propulsion. He applied for his first patent in January 1930, and submitted the idea to the Air Ministry, which declined it on the ground that practical difficulties in the way of development were too great. Firms approached were no more interested, but with some private assistance Whittle succeeded in coming to an arrangement which led to the formation of Power Jets, Ltd., in March 1936. Three months later, this company placed an order for the manufacture of a gas turbine unit with the British Thomson-Houston Co. of Rugby. Whittle was fairly confident in regard to the compressor and turbine elements but felt rather out of his depth with the combustion problem. Early tests proved that his fears regarding combustion were justified ; compressor performance, moreover, was far below expectations.

After the first series of tests at Lutterworth in 1937, the Air Ministry began to pay more attention to Whittle's work and by 1938 a reconstructed unit was ready for bench tests. After five hours' running severe damage resulted from a turbine blade failure. The Air Ministry had by this time accepted the fact that Whittle had the basis of a practicable aircraft power plant and a contract was placed with Power Jets for a ' flight ' engine—the W1 for installation in the Gloster E.28/39 aircraft. During manufacture certain major components were considered to be unairworthy, and it was decided to use these with some spare components, made for the first experimental engine, to build an ' early edition ' of the W1, known as the W1.X. The Gloster E.28, powered with the W1.X, left the ground for a short hop during taxying trials in April 1941, and flight trials with the W1 began during the following month. Designed for an air mass flow of 23 lb/sec, it realized a 21 lb/sec flow on test and developed 850 lb thrust at 16,500 r.p.m. The success attained gave a powerful stimulus to gas turbine research.

In the meantime, Power Jets had been authorized to go ahead with a more advanced engine, the W2, and the Gloster Aircraft Co. was requested to proceed with the design of a twin-engined interceptor fighter—the F.9/40, prototype of the Meteor. Moreover, direct contracts were placed with the British Thomson-Houston Co. and other firms for the manufacture and development of gas turbines, and Power Jets became a research and development organization supplying all other firms engaged with drawings and information.

Drawings of the ambitious W2 unit were handed over to the Rover Co for development, and when it was realized that the unit was extremely sensitive to design assumptions, it was changed by stages to bring it nearer to the standard of the W2B. A major and important change was later made by the adoption of straight-through combustion chambers in place of the reverse-flow type used hitherto. As a consequence of Ministerial policy, the development programme was later transferred to the Rolls-Royce Co., the Rover Co. being required to concentrate on other activities. Complete sets of drawings of the W2B were passed to the several firms by then engaged in gas turbine development. This unit was the prototype of the Rolls-Royce Welland, which subsequently powered the Meteor 1, and of the 'Type I,' the corresponding unit made by the General Electric Co. in the U.S.A.

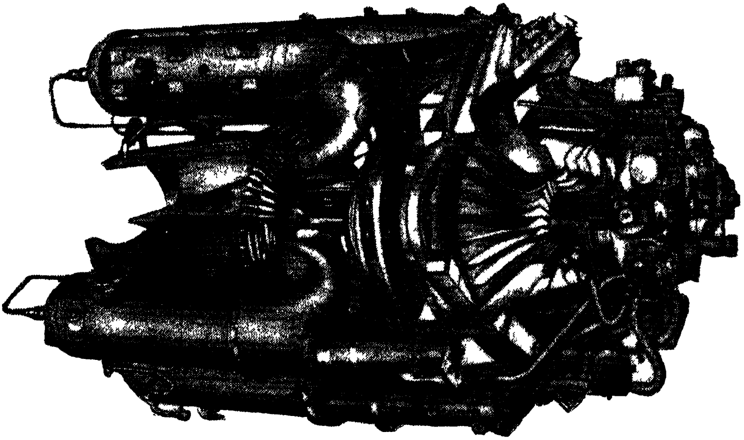


Fig. A : 1– A typical early Whittle unit, the Power Jets W2/700 with reverse-flow combustion chambers.

By arrangement between the British and American authorities, the W1.X turbine unit, a set of drawings of the W2B, and a small team of Power Jet engineers, were flown to America in the autumn of 1941, and development of the gas turbine at the G.E.C. works was initiated. The

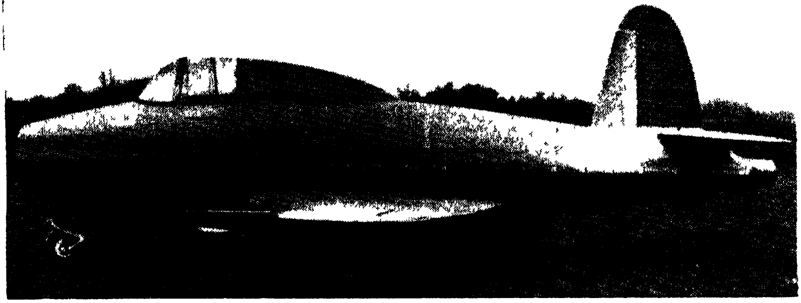


Fig. A 2—The first British turbojet, the Gloster E.28/39, powered by a Whittle unit made by the B.T.H. Co.

Ministry of Aircraft Production decided to plan for production of the W2B and of the Meteor, and many sceptics were converted The firms already engaged increased their activities considerably, Joseph Lucas Ltd commenced research on combustion and fuel systems, and firms which hitherto had practically ignored the gas turbine began to evince a lively interest.

In tracing the development of the axial-flow gas turbine in Great Britain it is necessary to refer to ' The Aerodynamic Theory of Turbine Design,' a paper prepared by Dr. A. A. Griffith at the Royal Aircraft Establishment in 1926. This paper indicated that the gas turbine was feasible as a means of aircraft propulsion and ten years later the Establishment obtained authority to build an axial-flow compressor, following

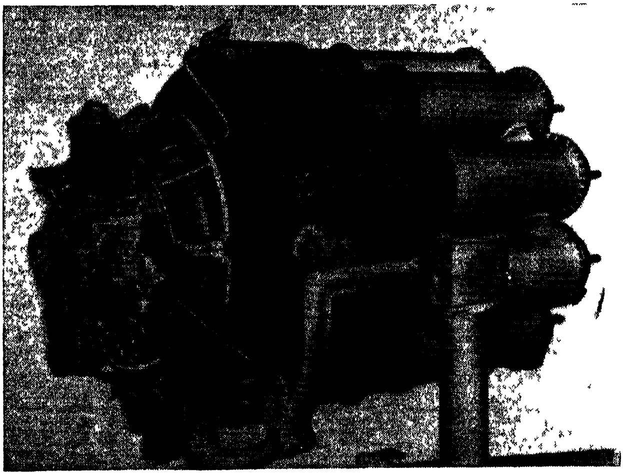


Fig. A : 3—Rolls-Royce Welland, the first of the River class turbojets.

its recommendations. In the following year, Mr. Hayne Constant, also of the R.A.E., concluded that a turbine to drive an airscrew could be constructed which would compare in specific weight and fuel consumption with the reciprocating engine. As the outcome of his work the R.A.E. collaborated with Metropolitan-Vickers in designing the B.10, a unit with a nine-stage axial compressor driven by a four-stage turbine.

A series of design studies followed, some embodying an annular combustion chamber and also an independent power turbine. By 1938 an experimental turbine compressor, resembling that suggested by Dr. Griffith, had been designed at the R.A.E. and was manufactured and tested by Armstrong Siddeley in 1939-40. Two years later Armstrong Siddeley was given a contract for a turboprop known as the A.S.X., the forerunner of the Python turboprop unit.

In January 1941, the de Havilland Co. commenced discussions on a turbojet unit for its new twin-boom DH.100 fighter, later to become famous as the Vampire. Three months later a decision was taken to commence working designs of a unit to develop a thrust of 3,000 lb. The first drawings were issued to the shops in August, and on April 13 in the following year the unit was ready for its first run on the test bed. In less than two months it was operating satisfactorily at its full designed thrust. Production of the jet engine outstripped that of the aircraft, and to obtain flight experience two units were first flown in a Gloster Meteor on March 5, 1943. It was not until September 20 that year that the Goblin was flown in a Vampire.

It is interesting to record that Goblin units were shipped to America and flown in the Lockheed Shooting Star. Both the Vampire and the Shooting Star by the spring of 1944 were exceeding 500 m.p.h., and it is claimed that these were the first aircraft in Britain or America to attain that speed.

In January 1945, the Goblin successfully passed an official type-approval test and now holds Type Certificate No. 1 for a turbine unit for aircraft propulsion.

With the close of the war a small design team at the Bristol Aeroplane Co. started gas turbine work. At that time no propeller-driving aircraft turbine had been designed, first because of the necessities of war, and secondly because a turbine to compete with the piston engine would need to be considerably more efficient than the jet engines of the war-time interceptor fighters. There were two approaches to this goal and both were beset with formidable development problems. If heat could be recovered from the turbine exhaust, a substantial improvement of efficiency could be had with the component efficiencies then attainable. The other approach was to use higher pressure ratios, for which a considerable raising of component efficiencies were prerequisite.

As the heat exchange problem appeared the more tractable at the time, the Theseus of 2,200 s.h.p. was designed and made. The compressor, having eight axial stages, followed by a single centrifugal stage, was driven directly by a two-stage turbine while power for the airscrew was furnished by an independently arranged single-stage turbine. Mounted to the rear of the turbines the heat exchanger had an annular tube matrix through which the efflux, slowed in a diffusing duct, passed axially. Air inlets and outlets were arranged round the periphery at the forward end. Overall dimensions of the exchanger were 45 in diameter, tapering to 36 in at the rear, and 31 in long.

The heat exchanger received 1,000 ft^3/sec of gas from the turbine at 500° C, and delivered 180 ft^3/sec of air to the combustion chambers at 300° C. It had a thermal ratio of 0·4, weighed 500 lb, and was calculated to save 150 lb of fuel per hour under typical cruising conditions.

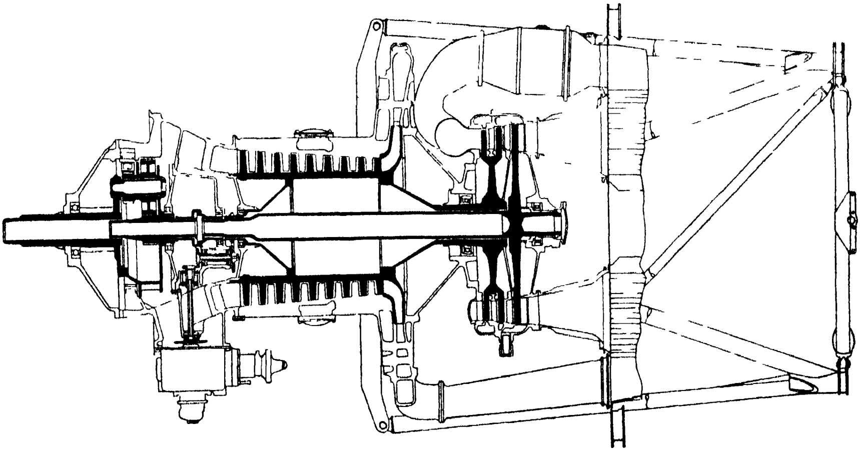


Fig. A : 4—Arrangement of Theseus turboprop with heat exchanger.

The turbine exhaust passed through 1,700 straight tubes of stainless steel, 0·012 in thick and 0·625 in diameter at the tube plates. By arranging the tubes along involute curves and flattening them, air passages of uniform width were formed between the adjacent involute rows. Air was thus taken from the eight inlet heaters, built inside the pressure-containing external drum, through groups of involute passages to the central space. This space formed a return header from which air returned to the peripheral space through other groups of involute passages alternating with the first-pass groups previously mentioned.

The thinness of the tubes and the close pitching made any form of mechanical or expanded point impracticable and the tubes were, therefore, brazed in place. The joints suffered during the hot starts that were a feature of those days. An alternative welded construction withstood the starts, but failed through contraction stresses on cooling.

It had become clear that the necessary improvements in component efficiencies could be realized and subsequent development has been concentrated in that direction, and towards exploiting the increased pressure ratios which can as a result be usefully employed. No other heat exchanger has ever been made for an aircraft turbine and present trends make it unlikely that heat exchangers will be used in any future designs.

Apart from its use for the study of heat exchanger construction and operation, the Theseus was also developed for operation without a heat exchanger. In this form it was successful in being the first turboprop unit in the world to complete a 100 hr M.o.S. type test and a 150 hr A.R.B. type test. It was also the first turbine unit, turboprop or turbojet, to complete a 500 hr endurance test under type-test conditions. Under Service conditions it was operated extensively in a Transport Command Lincoln flying regularly between Britain, Malta and Egypt.

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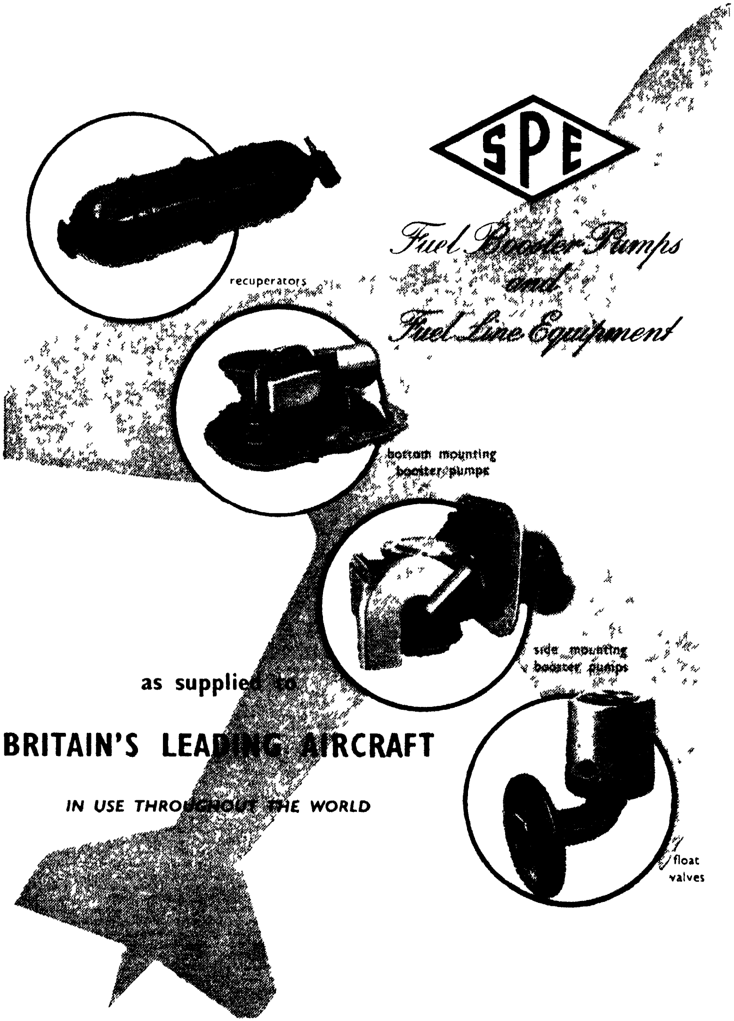

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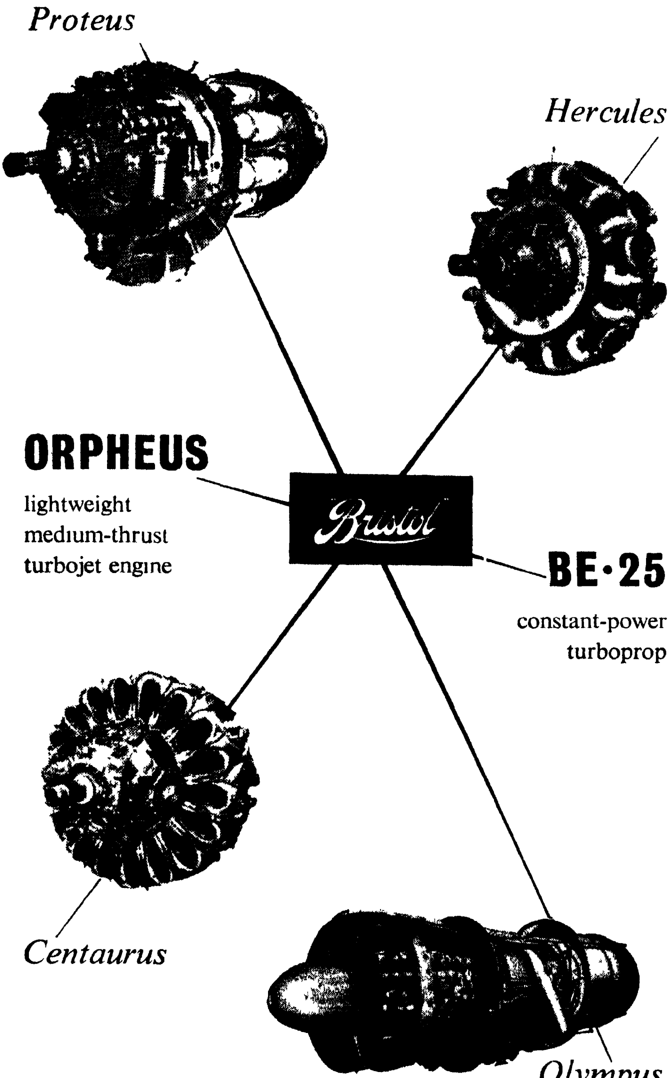

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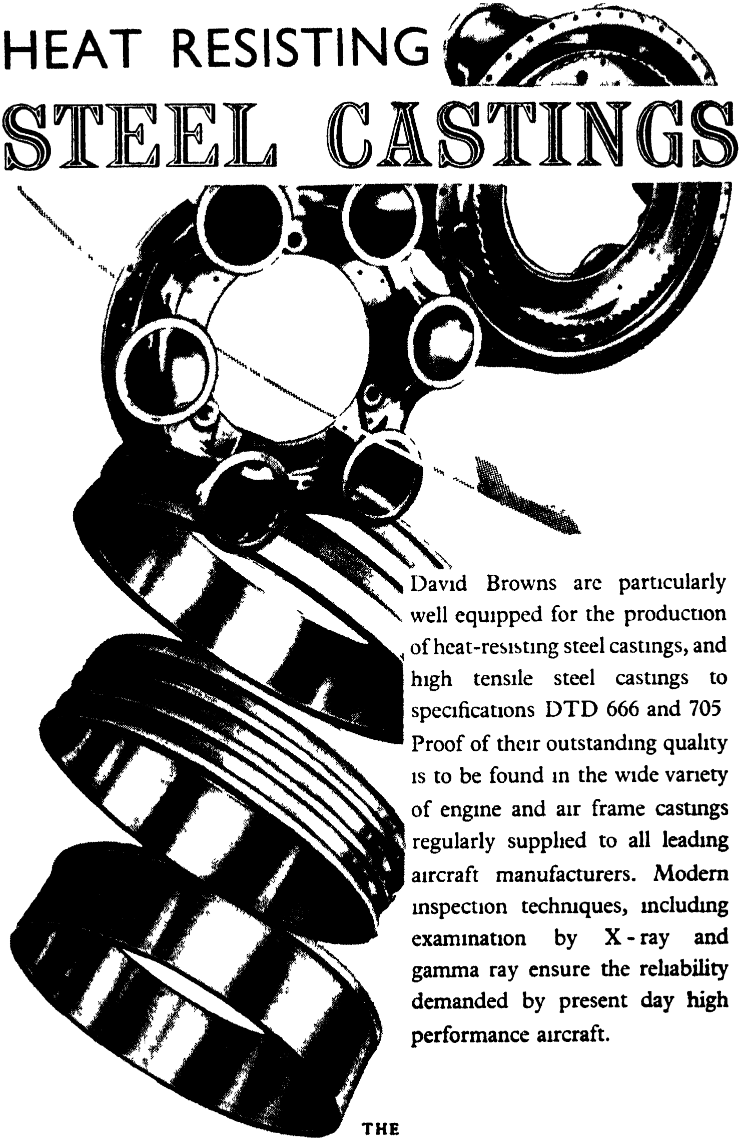

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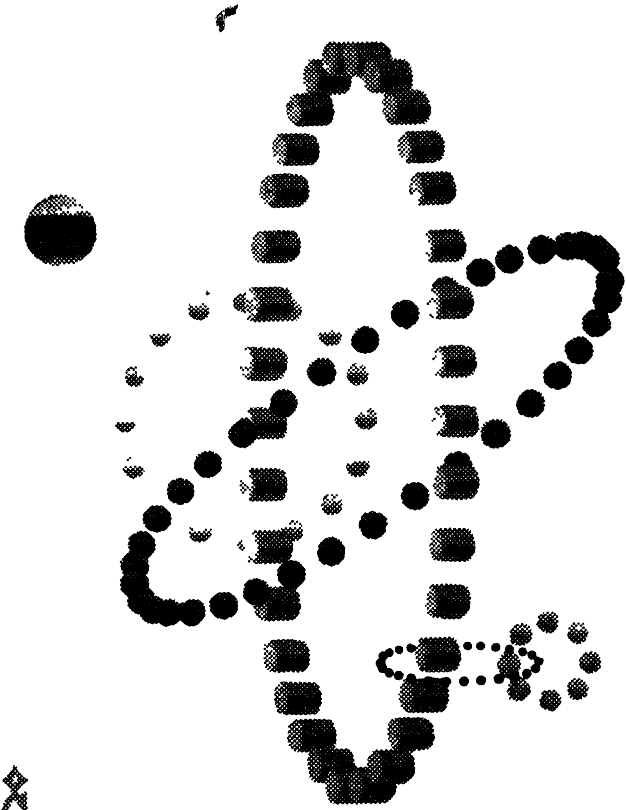

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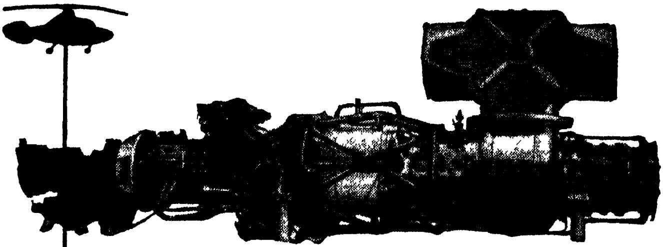

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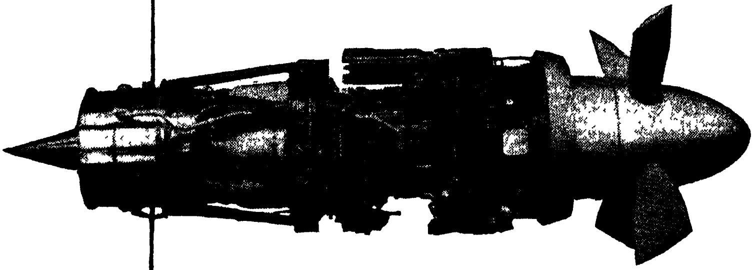

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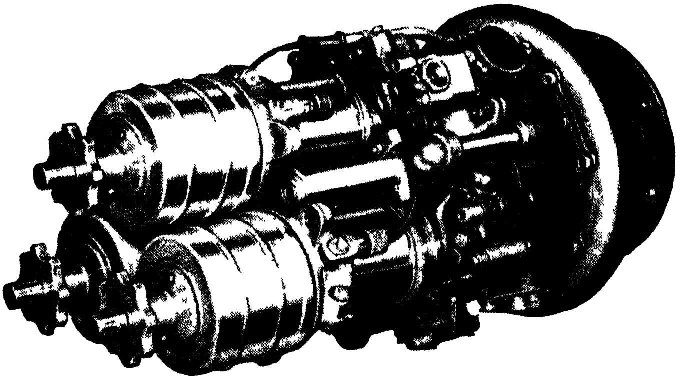

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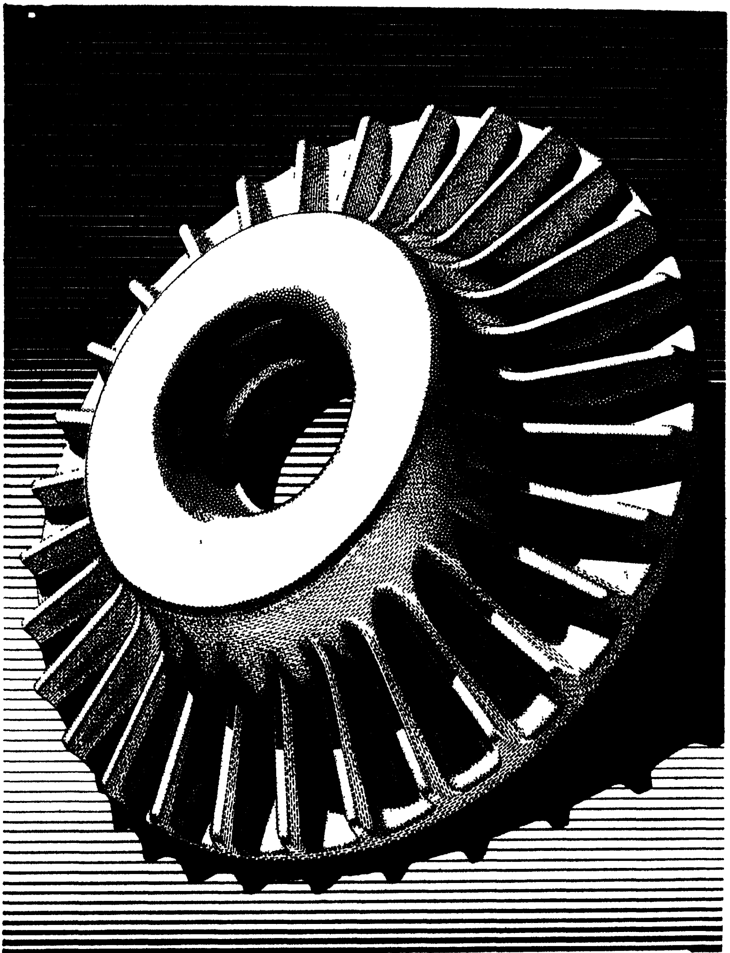

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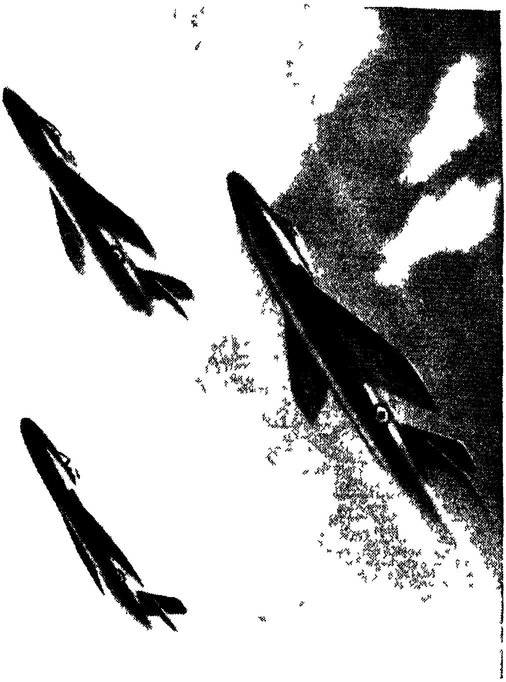

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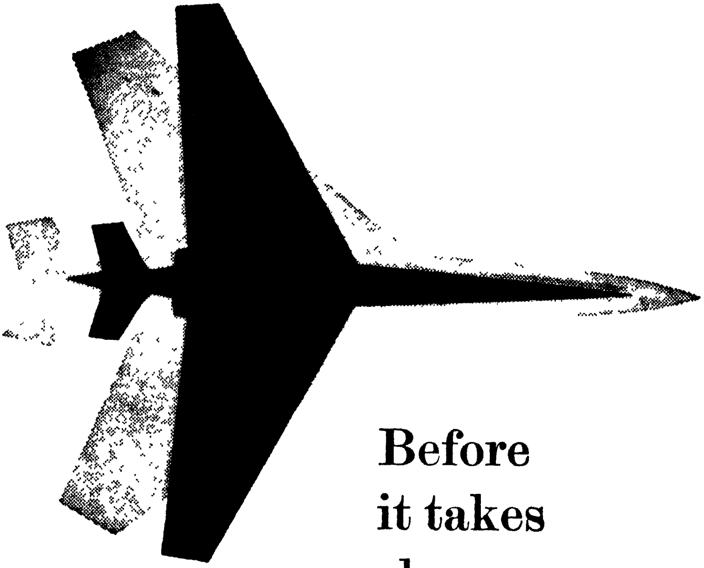

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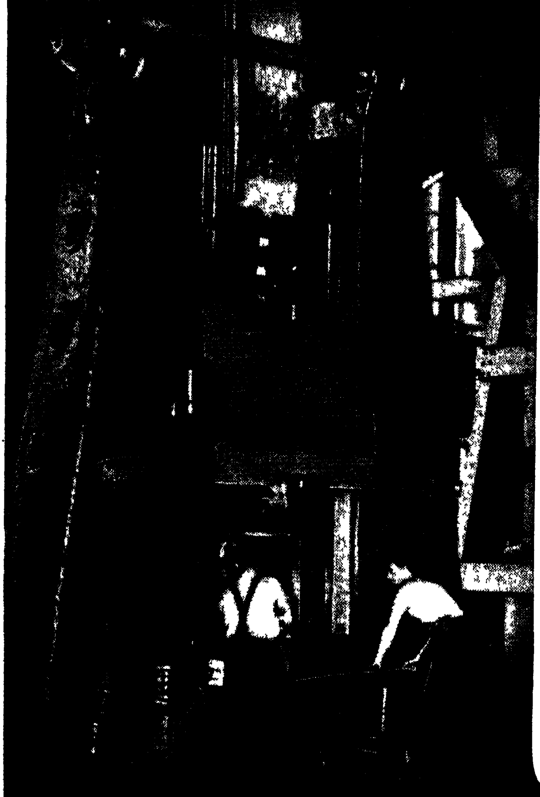

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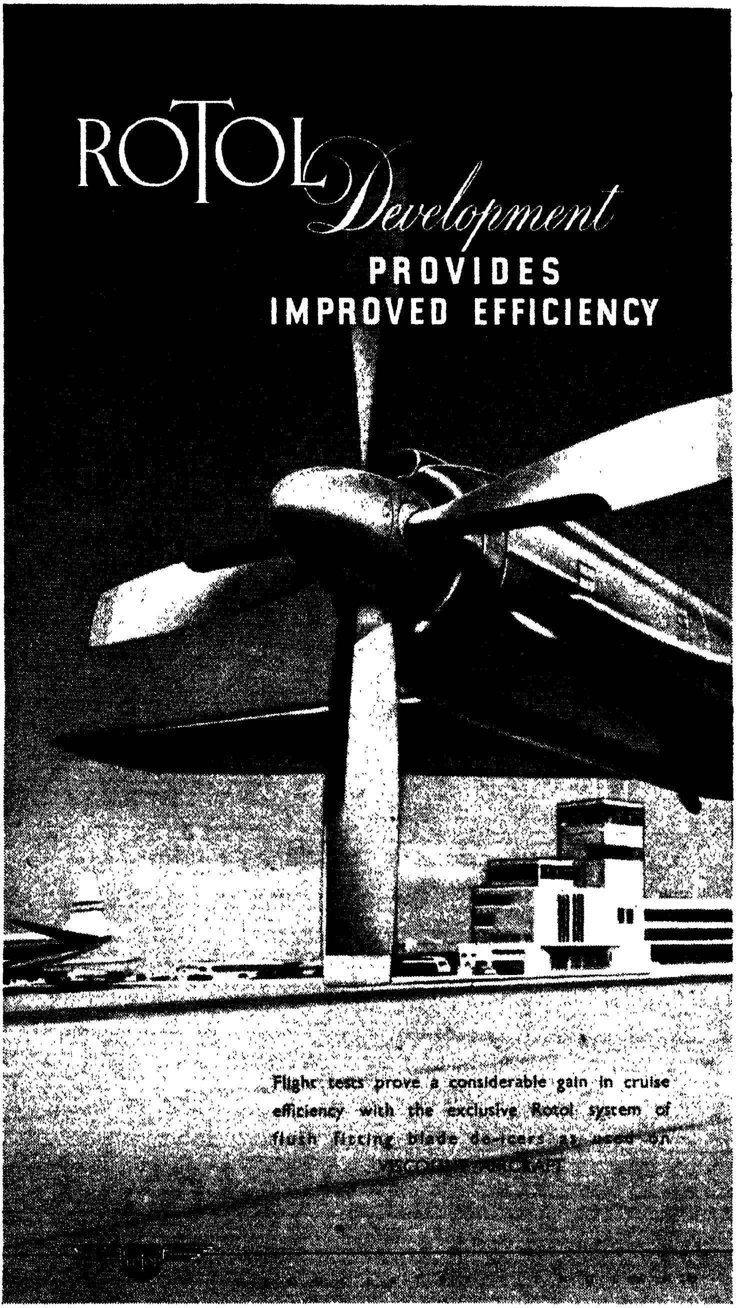

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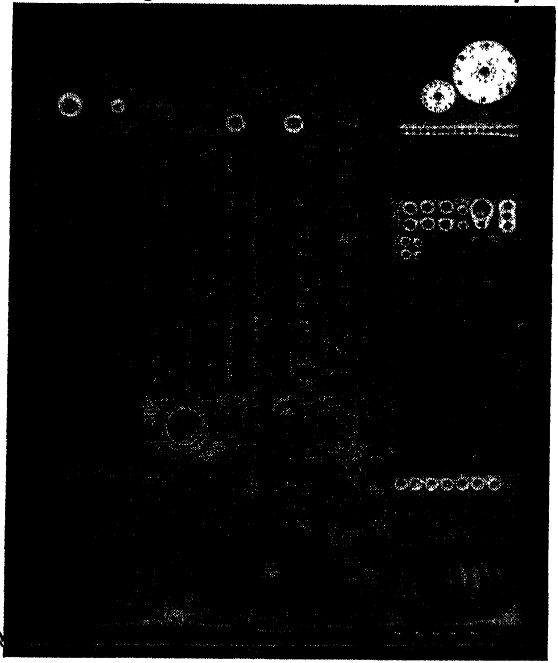

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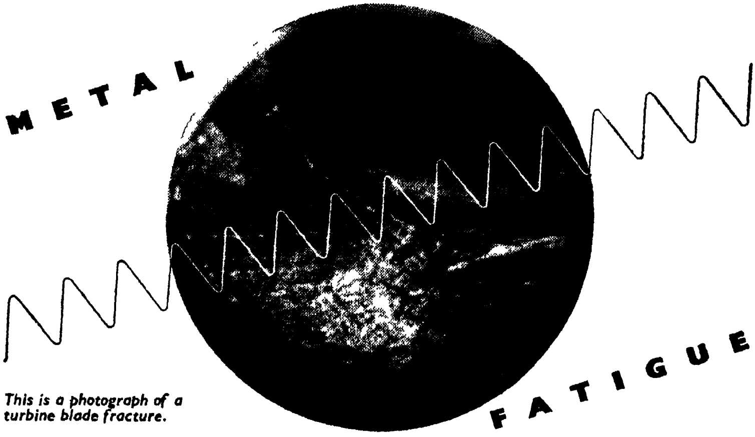

This is a photograph of a turbine blade fracture.

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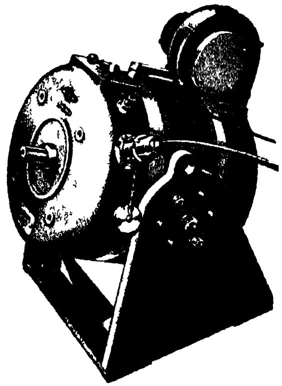

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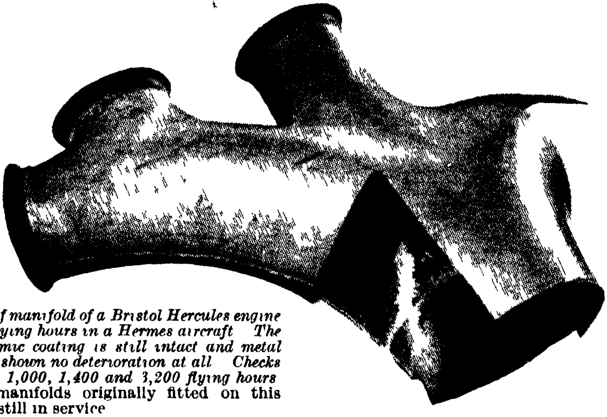

This is a half manifold of a Bristol Hercules engine after 3 200 flying hours in a Hermes aircraft The Ferro-Solaramic coating is still intact and metal analysis has shown no deterioration at all Checks were made at 1,000, 1,400 and 3,200 flying hours The other manifolds originally fitted on this aircraft are still in service

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